AD NUMBER
AD890753
LIMITATION CHANGES
TO: **Approved for public release; distribution is unlimited. Document partially illegible.**
FROM: **Distribution authorized to U.S. Gov't. agencies only; Test and Evaluation; DEC 1971. Other requests shall be referred to Air Force Rocket Propulsion Lab., Edwards AFB, CA 93523. Document partially illegible.**
AUTHORITY
AFRPL ltr 31 Jan 1974

AFRPL-TR-72-4

O_2/H_2 ADVANCED MANEUVERING PROPULSION TECHNOLOGY PROGRAM

ENGINE SYSTEM STUDIES FINAL REPORT

VOLUME I: AEROSPIKE ENGINE CONFIGURATION DESIGN AND ANALYSIS

Technical Report AFRPL-TR-72-4

Rocketdyne
A Division of North American Rockwell Corporation
6633 Canoga Avenue
Canoga Park, California

December 1971

Distribution limited to U.S. Government Agencies only; data based on test and evaluation; December 1971. Other requests for this document must be referred to AFRPL (STINFO), Edwards, California 93523.

Air Force Rocket Propulsion Laboratory
Air Force Systems Command
United States Air Force
Edwards Air Force Base, California 93523

AFRPL-TR-72-4

O_2/H_2 ADVANCED MANEUVERING PROPULSION TECHNOLOGY PROGRAM

ENGINE SYSTEM STUDIES FINAL REPORT

VOLUME I: AEROSPIKE ENGINE CONFIGURATION DESIGN AND ANALYSIS

Technical Report AFRPL-TR-72-4

Rocketdyne
A Division of North American Rockwell Corporation
6633 Canoga Avenue
Canoga Park, California

December 1971

Distribution limited to U.S. Government Agencies only; data based on test and evaluation; December 1971. Other requests for this document must be referred to AFRPL (STINFO), Edwards, California 93523.

Air Force Rocket Propulsion Laboratory
Air Force Systems Command
United States Air Force
Edwards Air Force Base, California 93523

FOREWORD

This technical report presents the results of the Engine System Studies for the Aerospike Engine Configuration Design and Analysis conducted as part of the O_2/H_2 Advanced Maneuvering Propulsion Technology (AMPT) Program. The studies were conducted by the Rocketdyne division of North American Rockwell during the period 1 January 1971 to 1 October 1971 as part of United States Air Force Rocket Propulsion Laboratory Contract F04611-67-C-0116.

The Air Force Program Manager was Mr. W. W. Wells. Mr. H. G. Diem was the Rocketdyne Program Manager. For the Aerospike Engine Task, Mr. D. H. Huang was the Study Manager and Mr. D. B. Wheeler was the Principal Engineer.

This report, Rocketdyne report R-8807, consists of three volumes:

- Volume I: Single-Panel and Double-Panel Aerospike Engine System Studies

  Volume II: Alternate Engine System Studies

  Volume III: 10,000-Pound-Thrust Bell Engine System Studies

Volumes I and II were submitted on 15 November 1971.

This technical report has been reviewed and approved.

W. W. Wells
AFRPL AMPT Program Manager
RPRES

ABSTRACT

The engine system design and analysis studies provide a detailed definition of two 25,000-pound-thrust O_2/H_2 aerospike engines. The single-panel aerospike engine design point corresponds to the demonstrator thrust chamber configuration, specifically, chamber pressure and area ratio equal to 750 psia and 110:1, respectively. A second engine system and component design and operational description also is provided for the selected optimum aerospike engine employing a double-panel thrust chamber cooling circuit. The double-panel aerospike engine design has a chamber pressure and area ratio of 1000 psia and 200:1, respectively. These engine systems are designed to provide 5:1 throttling and off-design mixture ratio operation. The study effort also included the effects of variations in certain design parameters on engine performance, weight, propellant flow balances, life capability, development time and cost, and maintenance requirements. Additional parametric information is provided for design thrust levels between 8,000 and 50,000 pounds.

CONTENTS

ILLUSTRATIONS

TABLES

ACKNOWLEDGMENTS

The work reported in this volume represents the concerted effort and expertise of many members of the Rocketdyne organization. Contributions of major significance were made by the following personnel:

H. C. Wieseneck	- Technical Review
W. M. Stanley	- Engine System Analysis
D. J. Levack	- Preliminary System Design, Engine System Analysis
W. H. Blendermann	- Injector and Thrust Chamber Design and Analysis
G. Allen	- Engine System Design
J. Shoji	- Heat Transfer Analysis
A. Csomor	- Turbopump Configuration Selection and Analysis
J. M. Zorad	- Turbopump Design
H. Gardy	- Turbine Analysis
R. L. Nelson	- Dynamic Analysis
T. Young	- Stress and Weights Analysis
A. T. Zachary	- Program Plans
M. S. Bensky	- Ignition System Selection
W. S. Eierman	- Cost Analysis, Failure Mode Effects
E. H. Cross	- Final Report Preparation
J. A. Barrett	- Editor

NOMENCLATURE

AAN^2	Turbine annulus area x speed squared, $in.^2$-rpm^2
A_N	Turbine nozzle area, $in.^2$
C_M	Pump inlet velocity, ft/sec
C_o	Turbine gas spouting velocity, ft/sec
D_{imp}	Pump impeller diameter, inches
D_M	Turbine pitch diameter, inches
DN	Pump shaft diameter x speed, mm-rpm
F	Engine thrust, pounds
g	Gravitational constant
H	Head, feet
I_s	Specific impulse, seconds
MR	Mixture ratio (oxidizer/fuel)
N	Speed, rpm
N_S	Specific speed, rpm $gpm^{1/2}/ft^{3/4}$
NPSH	Net positive suction head, feet
P	Pressure, psia
P_c	Chamber pressure, psia
P_d	Pump discharge pressure, psia
PR	Turbine pressure ratio
Q	Flowrate, gpm
R	Gas constant
S_S	Suction specific speed, rpm $gpm^{1/2}/ft^{3/4}$

T	Temperature (F or R as specified)
U	Turbine pitch line velocity, ft/sec
U_{imp}	Pump impeller tip speed, ft/sec
W	Weight, pounds
$\dot{W}$	Flowrate, lb/sec
δ	Nozzle percent length
Δ	Difference
ϵ	Nozzle area ratio or turbine percent admission
ϵ_c	Chamber contraction ratio
η	Efficiency
σ	Standard deviation
τ	Secondary flow ratio, $\dot{W}_s/\dot{W}_p$

INTRODUCTION

A 9-month system design and analysis study was performed to define oxygen/hydrogen main rocket engine designs applicable to high performance space vehicles such as the high energy upper stage or the orbit-to-orbit shuttle space vehicle. In this study, two aerospike engine thrust chamber design approaches were considered: single-panel and double-panel regenerative cooling jackets. This report (Volume I) describes the results of the aerospike engine system investigations including configuration selection, design description, supporting analyses, and cost data, and assesses the impact of changing various design requirements.

In the study, pump-fed engine systems in the thrust range of 8,000 to 50,000 pounds thrust were investigated. The engines had a nominal mixture ratio of 5.5:1 and were required to operate over a range of ±0.5 mixture ratio units and to be throttleable to 20-percent thrust. Additionally, the engine was required to be maintenance-free for 2 hours or 60 thermal cycles, to be inspected without major overhaul four times, and to be overhauled four times within a total engine life-cycle of 1500 thermal cycles and 50 hours of operation. Vehicles were returned to the ground after each flight and engine maintenance was, therefore, ground based. Inspection and overhaul cost guidelines were specified.

Two aerospike engine designs are discussed in detail in this report. The first design, called single-panel, because it uses only the fuel as a regenerative coolant, has an area ratio of 110 to 1 and a maximum chamber pressure of 750 psi. This design point corresponds exactly to the single-panel thrust chamber demonstrator hardware being fabricated and tested under other tasks on this program. Some additional performance could be obtained with the single-panel design by enlarging the nozzle area ratio to the maximum possible value of 150 to 1 at the same chamber pressure. However, the more conservative expansion ratio was selected to provide an additional operating safety margin for the demonstration hardware.

The second aerospike design is called double-panel because both fuel and oxidizer are used as regenerative coolants in the combustion section to provide additional

cooling capability. The optimum double panel has a chamber pressure of 1,000 psi and a nozzle expansion ratio of 200:1. This design point defines the maximum possible performance for the aerospike concept at a thrust level of 25,000 pounds. Demonstrator hardware with slightly more conservative operating conditions (950-psi chamber pressure and 190:1 expansion) is being built and tested under separate parts of this program.

The study of the two engine systems was divided into five areas of investigation. Engine data for the entire thrust range and a variety of engine configurations were developed under the Engine Design Data for Different Thrust Levels Study section. These data are reported in this report. In the 25,000-Pound-Thrust Engine Design Study section, a detailed engine system design effort was carried out for a selected engine configuration. Results were summarized in an "Engine Design Description" report (Ref. 1) and documented completely in the current volume.

The effects of changing the design requirements for the design point engine were explored in the 25,000-Pound-Thrust Varying Design Conditions section. The results are reported in this volume. In the Mixture Ratio and NPSH Study section, the effect of varying design mixture ratio and NPSH on engines of 8,000, 15,000, 25,000 and 50,000 pounds thrust were investigated. Demonstrator engine, engineering development program, and first production unit schedules and costs for the point design engine are reported in this volume.

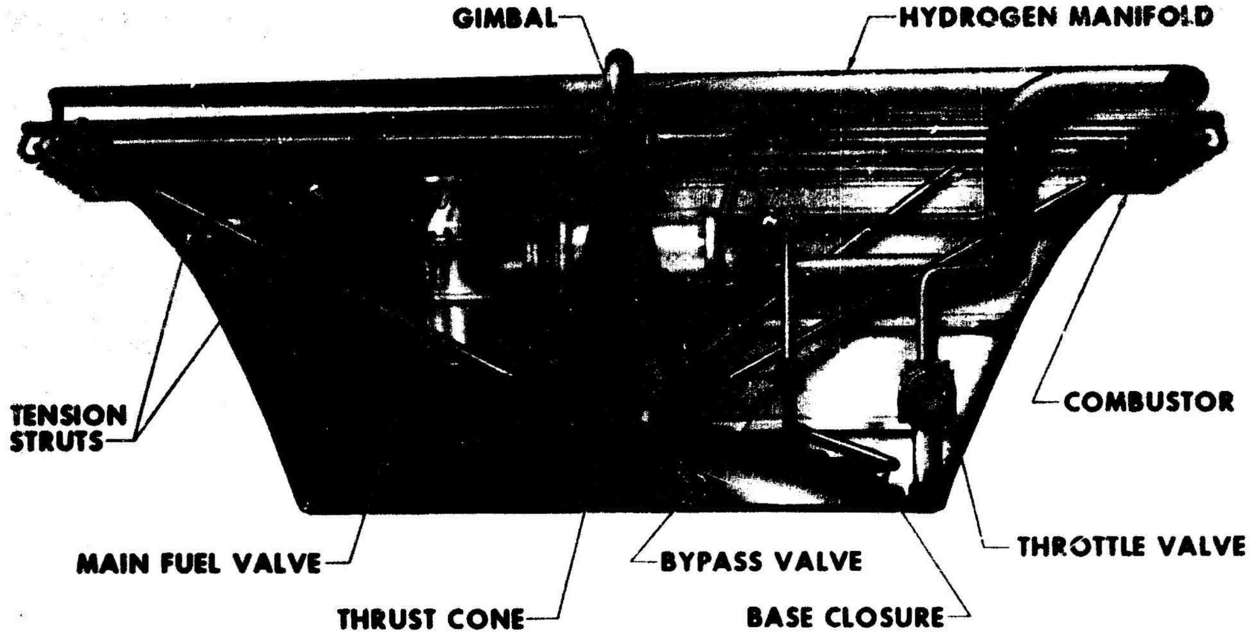
This study was part of one task in the Advanced Maneuvering Propulsion Technology Program (Contract F04611-67-C-0116) that includes advanced development effort on the aerospike thrust chamber, with both single-panel and double-panel regenerative cooling circuits, and studies of alternate O_2/H_2 engine system configurations. The results of the other tasks are presented in separate volumes. Because aerospike thrust chamber test information will continue to be accumulated during the advanced development effort, the engine system design may change slightly. The Engine Design Description report (Ref. 1) will, therefore, be revised quarterly to include the effect of these test data on the engine design.

SUMMARY

An engine system design and operational definition study was completed for an advanced rocket engine for space systems applications. Two engine system designs were completed: a single-panel chamber design and a double-panel chamber design. The two 25,000-pound-thrust aerospike engine system designs differ in the thrust chamber cooling circuit, design chamber pressure, and area ratio. A summary of the basic design parameters for the two engines is presented in Fig. 1. The engine configurations consist of a 24-segment aerospike thrust chamber with a truncated spike expansion nozzle. The propellants are pump-fed to the thrust chamber, and the engine is capable of 5:1 throttling. The design point for the single-panel engine is near optimum and was selected to correspond to the demonstrator thrust chamber design. The double-panel engine design point was selected based on the results of an optimization analysis. Both engine systems employ an expander topping cycle for turbopump power based on a cycle selection study. The heated hydrogen, after passing through the regenerative cooling jacket, is expanded through the turbines before it is injected into the main combustion chamber.

Centrifugal-type pumps which are directly driven by axial flow impulse turbines in a parallel flow arrangement are used. The segmented combustion chamber is constructed of individual segment liners fabricated from cast NARloy.* Two continuous structural rings, fabricated from titanium alloy, form the inner and outer body support to the segment liners. A single-pass nozzle of tubular wall construction is cooled in series with the segments. Identical control systems were adopted for both engine systems, and the flow circuitry is the same except for the regenerative coolant flow paths. A schematic of the double-panel engine is shown in Fig. 2. The control system consists of main propellant valves located immediately upstream of the respective turbopumps, an oxidizer turbine inlet control valve, and a turbine bypass valve. The turbine bypass control valve is a variable area type and provides engine thrust control by varying the heated hydrogen flowrate through the turbines. The oxidizer turbine inlet flow control valve also is a variable area type and provides engine mixture ratio control.

*Low silver alloy of copper



	SINGLE-PANEL DESIGN	DOUBLE-PANEL DESIGN
VACUUM THRUST, POUNDS	25,000	25,000
MAXIMUM CHAMBER PRESSURE, PSIA	750	1000
EXPANSION AREA RATIO	110:1	200:1
NOMINAL ENGINE MIXTURE RATIO	5.5:1	5.5:1
MIXTURE RATIO OPERATING RANGE	5.0:1 TO 6.0:1	5.0:1 TO 6.0:1
THRUST THROTTLE RATIO	5.0:1	5.0:1
VACUUM SPECIFIC IMPULSE AT 5.5:1 MIXTURE RATIO, SECONDS	458	471
DRY ENGINE WEIGHT, POUNDS	360	398
GIMBAL ANGLE, DEGREES	±5	±5
ENGINE LENGTH, INCHES	24.1	27.0
ENGINE DIAMETER, INCHES	61.6	66.9

Figure 1. 25K O_2/H_2 Aerospike Engine

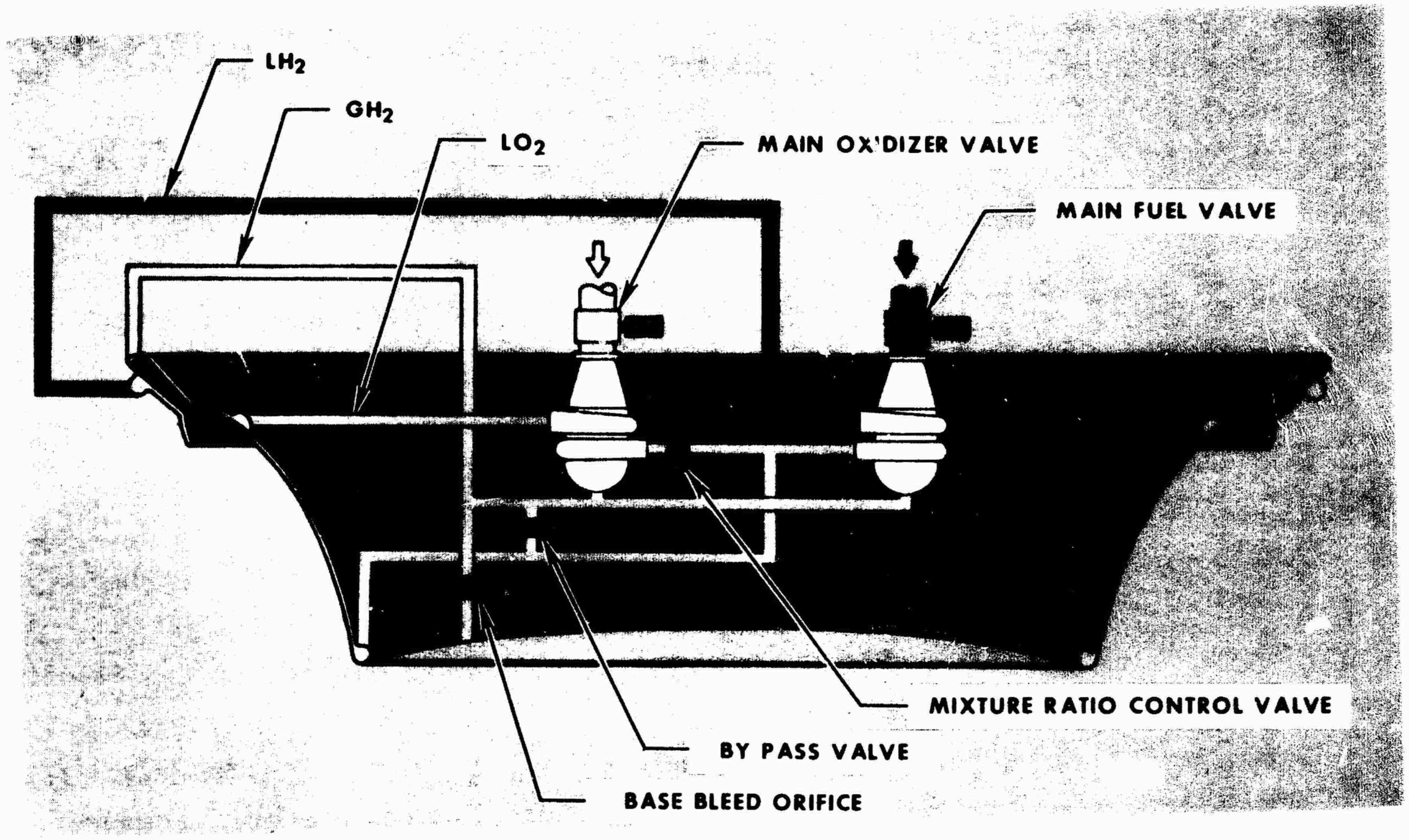


Figure 2. Double-Panel Aerospike Engine System Flow Schematic

Analysis of the engine components and control system provided for a nominal turbine bypass flow at full-thrust operation. This control allows a turbopump drive cycle power margin to accommodate possible variations in the predicted pressure schedules and turbopump efficiencies that may be experienced during the development program.

The engine, designed for multiple starts at altitude, utilizes a tank-head power start sequence. The fuel is fed through the regenerative cooling jacket under tank pressure where it picks up residual heat. The warm hydrogen then passes through the turbines providing the initial power to the pumps. After ignition, the engine bootstraps itself to full power. A combustion wave ignition system was selected where a central mixing chamber distributes the premixed propellants to each at the combustion chamber segments. A spark igniter then ignites the mixed propellants and a combustion wave travels to each segment.

Engine system start, cutoff, and transient operation and control methods were analyzed and established for the two engine systems. A failure modes and effects analysis was conducted and recommendations are made to ensure a fail-safe engine system.

Demonstrator engine and engine development program plans and costs were established for these baseline engine systems. Engine development program costs are minimized by overstress testing and failure-mode detection during a thorough component level test effort. In this way, the number of engine system tests is reduced.

A design point variation analysis was conducted for the two-engine system designs to determine the influence of certain specified design requirements on the major engine operational parameters, design features, maintenance, program plans, and cost. A matrix of dependent and independent parameters was specified by the contract.

Parametric engine information in the form of specific impulse, weight, and dimensions were generated and are presented in this report for design thrust levels between 8,000 and 50,000 pounds. Variations in turbine drive cycle, design chamber pressure, area ratio, and mixture ratio are included. Engine system optimization studies were conducted and design point definition was established for various thrust levels within the parametric thrust range.

For selected thrust levels within the specified range, variations in the nominal design values of mixture ratio and pump inlet NPSH were investigated to determine the effects on engine operation and design features.

TURBINE DRIVE CYCLE SELECTION

INTRODUCTION

The turbopump drive cycle selection consisted of an initial screening of the candidate cycles with a selection of the two highest ranked cycles for a more detailed comparison. Also, both a parallel turbine arrangement and a single turbine with a direct-drive fuel pump and a gear-driven oxidizer pump were compared, thus making a total of four systems that were carried through the more detailed evaluations. Series turbines were not examined because they produce delivered engine performance equal to the single turbine or between the single and parallel turbines, and have complexity and system flexibility characteristics that also are between those of the single and parallel turbines. A single-turbine/single-shaft arrangement would require a low-pressure pump on the oxidizer side to compare faborably at this thrust level because the turbine speed and fuel pump speed would otherwise be forced down to that required for the main oxidizer pump to satisfy the NPSH requirement. Thus, performance, flexibility, and complexity were bracketed by considering only the single and parallel turbine arrangements.

The cycle comparisons were conducted using a preliminary version of the 25,000-pound-thrust, single-panel aerospike engine at the design point specified in the contract. A second 25,000-pound-thrust engine system design point was established for a double-panel aerospike engine later in the study effort. The detailed turbine drive cycle comparison was not repeated for the double-panel engine design point because it was obvious from the parametric engine information and subsequent engine optimization studies that the relative comparisons between the candidate cycles were essentially equal for both the single-panel and double-panel engines. On this basis, the comparisons and conclusions presented for the single-panel engine system are considered to be equally valid for the double-panel engine.

DISCUSSION

For each of the six cycles considered in the initial screening, a complete preliminary system was examined at the nominal operating point of 25,000-pounds-thrust, 750-psia chamber pressure, and 5.5:1 mixture ratio. A summary of the engine design point definition is shown in Table 1 . Delivered performance was calculated at the nominal point and for a ±0.5 engine mixture ratio excursion. Performance also was calculated for a throttled operating condition with a chamber pressure of 150 psia and engine mixture ratios of 5.0, 5.5, and 6.0:1.

TABLE 1. O_2/H_2 AEROSPIKE ENGINE DESIGN POINT DEFINITION FOR TURBINE DRIVE CYCLE COMPARISON

DESIGN THRUST	25,000 POUNDS
CHAMBER PRESSURE	750 PSIA
AREA RATIO	110:1
NOMINAL ENGINE MIXTURE RATIO	5.5:1
ENGINE MIXTURE RATIO OPERATING RANGE	5:1 to 6:1
THROTTLE RATIO	5:1
SERVICE LIFE BETWEEN OVERHAULS	
DURATION	10 HOURS
CYCLES	300
TOTAL LIFE	
DURATION	50 HOURS
CYCLES	1500
FUEL PUMP NPSH	60 FEET
OXIDIZER PUMP NPSH	16 FEET

Each of the candidate cycles, followed by details of the analytical methods employed and the component performance analysis, are presented in this report. Following

these, the major engine operating characteristics for all cycles and turbine arrangements are presented. A preliminary selection of two of the most promising cycles is made and a more detailed evaluation is then made leading to a final selection.

CANDIDATE CYCLES

Six engine drive cycles, as shown in Fig. 3 and 4, were considered as possible candidates for the AMPT engine:

Expander Topping

Staged-Combustion Topping

Gas Generator

Thrust Chamber Tapoff

Hydrogen Bleed

Auxiliary Heat Exchanger

The first two cycles are referred to as "closed" cycles because the entire turbine flow can be included in the main chamber flow. Only a small portion (<0.5 percent) is directed through the nozzle base and was established to maximize base pressure thrust contribution. The last four are referred to as "open" cycles because the turbine or secondary flowrate is completely determined by turbomachinery parameters. All of the turbine flow is directed through the nozzle base and is always greater than the optimum base flow for base pressure augmentation.

Expander

In the expander cycle, turbine power is derived from running most of the hot hydrogen flow from the cooling jacket through low-pressure ratio turbines (5 to 20 percent is normally bypassed as a control reserve). The turbine flow is then combined with the bypass flow. A small amount of the hydrogen (0.2 percent) is

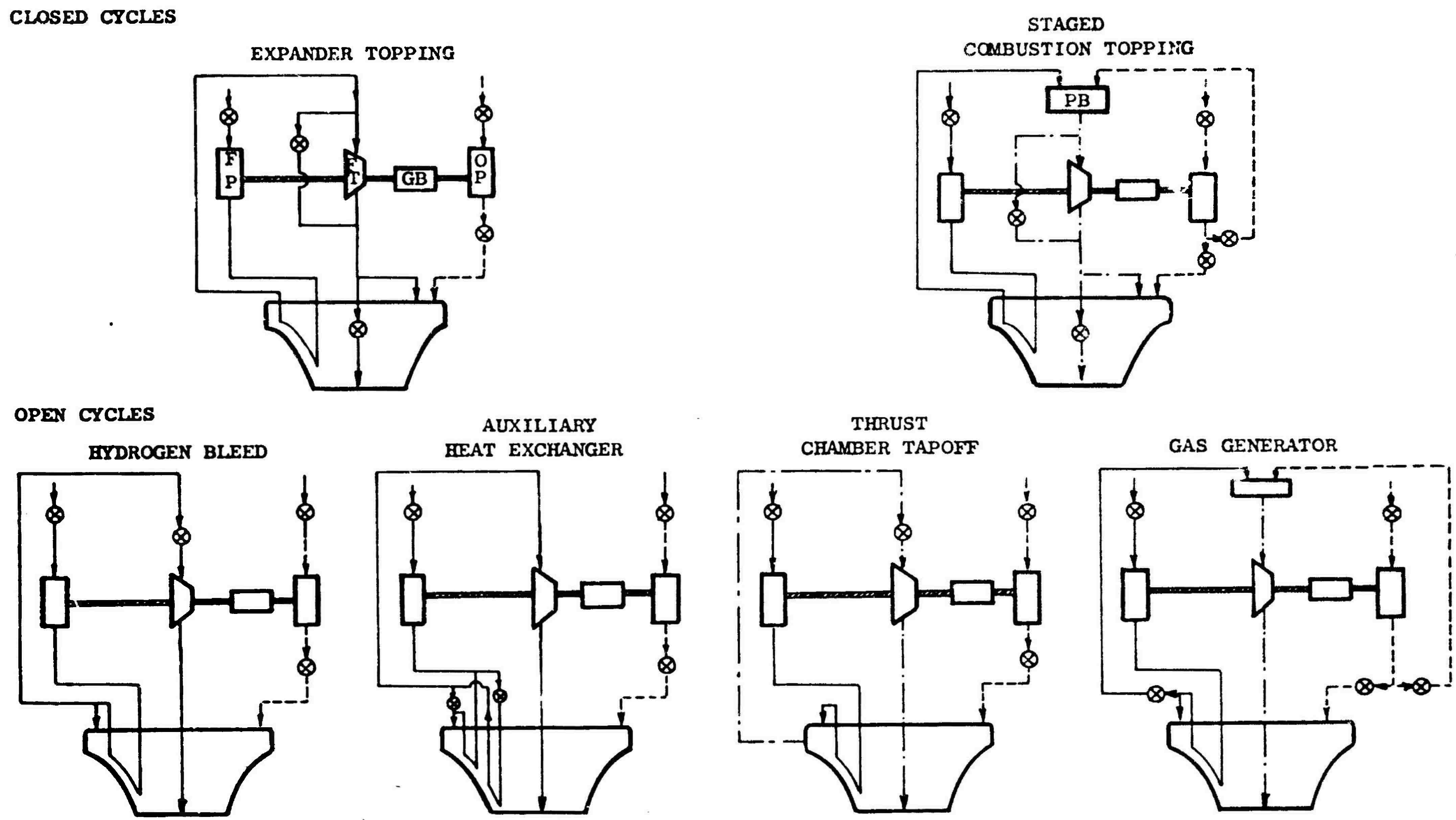


Figure 3. Engine Cycle Schematics-Single Turbine

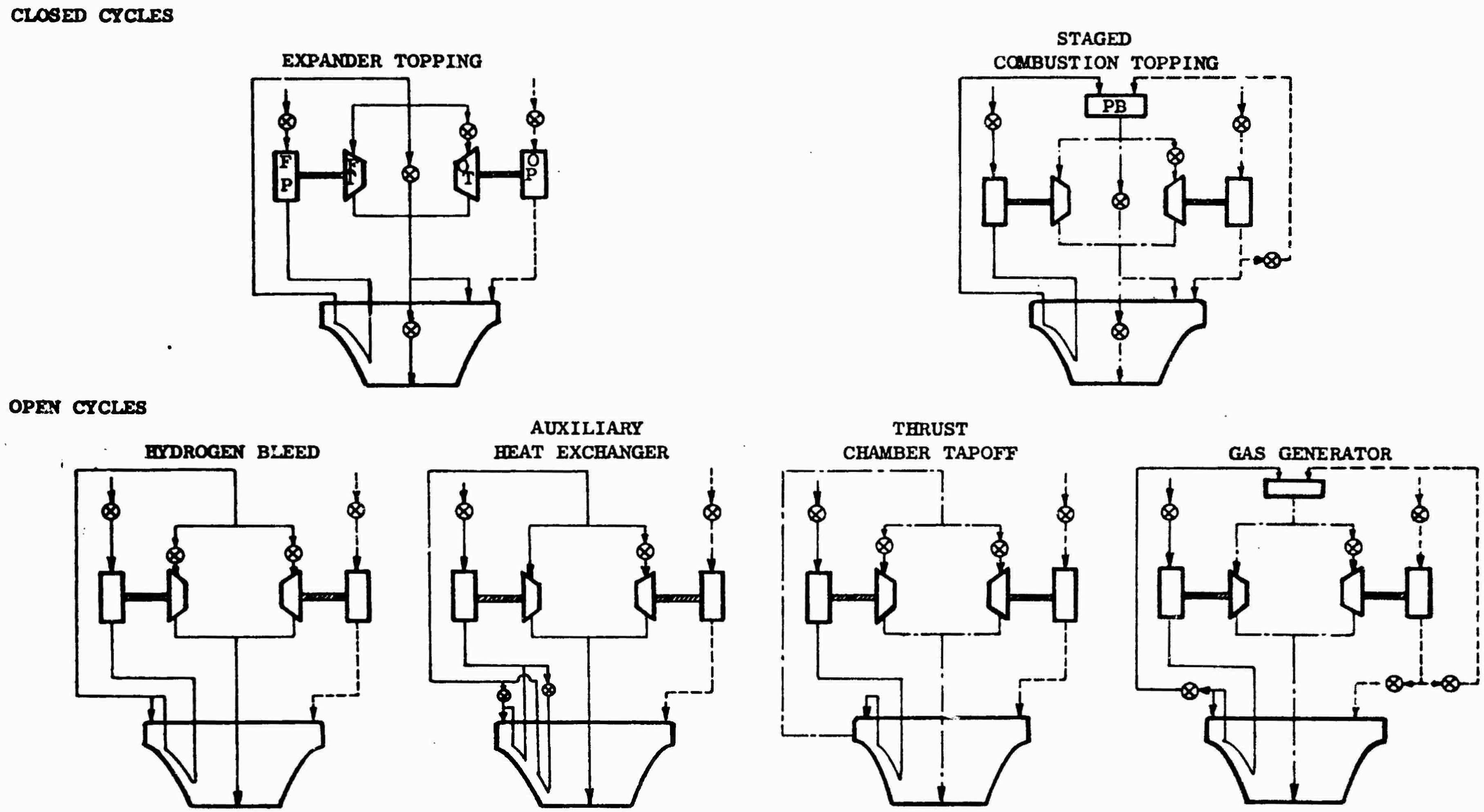


Figure 4. Engine Cycle Schematics-Parallel Turbines

then passed to the base to achieve optimum base pressure and the rest is injected into the main chamber. A cycle schematic is shown in Fig. 3 for a single turbine arrangement, and in Fig. 4 for a parallel turbine arrangement.

A variation on the expander cycle was investigated and referred to as the augmented expander cycle. The augmented expander cycle is similar to the expander except that a small amount of oxygen is injected into the hydrogen stream between the cooling jacket exit and the turbine inlet to add energy to the turbine drive flow. The cost is more complexity (i.e., some means to inject the oxygen) and increased oxygen pump discharge pressure. Predicted performance of this cycle is within 0.1 second of delivered specific impulse of the expander cycle. Thus, at conditions where there is sufficient energy in the cooling jacket exit hydrogen flow to power the turbines, the use of an augmented expander cycle can provide nothing except additional complexity and development effort. This is not to say that the augmented expander cycle has no application; its use is to extend the range of closed-cycle operation to regimes (i.e., higher pressure or higher thrust) where the heated hydrogen flow is not energetic enough to power the turbines. Very little difference exists between the staged-combustion cycle and the augmented expander cycle. At this point, the augmented expander cycle is seen as a method of injecting a small amount of oxidizer directly into the turbine inlet duct. In actual practice, this will probably require an injector, mixing chamber, and igniter which will differ very little from a precombustor.

Staged Combustion Topping

The staged combustion topping cycle is very similar to the augmented expander cycle. The only difference is in the mechanical method of introducing the oxygen into the hydrogen flow. The staged combustion topping cycle uses a precombustor or gas generator to introduce the oxygen into the turbine flow with consequent higher required fuel and oxidizer pump discharge pressures. This cycle provides nothing but additional complexity when an expander cycle will provide adequate turbine power. However, this cycle is capable of extending the range of closed-cycle operation very far beyond the capabilities of a simple expander cycle. A cycle schematic is shown in Fig. 3 for a single turbine arrangement, and in Fig. 4 for a parallel turbine arrangement.

Gas Generator

In the gas generator cycle, both hydrogen and oxygen in the amount required (~2 percent of total flow at a mixture ratio of ~0.7:1) is routed from the thrust chamber cooling jacket, combusted in a gas generator, and used to power high-pressure ratio turbines. The mixture ratio is limited by the maximum allowable turbine inlet temperature. As the maximum allowable temperature is decreased, the total hot-gas flow must be increased. The turbine exhaust flow is directed to the aerospike nozzle base where it increases the base pressure and total engine thrust. In nearly every case, the required turbine flowrate is greater than the optimum base flowrate. However, the resulting engine performance loss is small (~0.5 percent or less). A cycle schematic is shown in Fig. 3 for a single turbine arrangement and in Fig. 4 for a parallel turbine arrangement.

Thrust Chamber Tapoff

The thrust chamber tapoff cycle is similar to the gas generator cycle in concept, except that the source of the turbine drive gas is the main chamber. Hot gas from the combustion chamber is bled off mixed with cold H_2. The tapoff gases are directed through the high-pressure-ratio turbines and exhausted into the nozzle base. A cycle schematic is shown in Fig. 3 for a single turbine arrangement and in Fig. 4 for a parallel turbine arrangement.

Hydrogen Bleed

The hydrogen bleed cycle obtains its turbine drive gas by extracting the required amount of hot hydrogen from the cooling jacket exit. The hydrogen bleed flow is directed through the high-pressure-ratio turbines and exhausted through the nozzle base which usually results in above-optimum aerospike nozzle base flow and this excess flow from the turbine exhaust cannot be injected into the chamber due to the pressure difference. When the cooling jacket exit temperature is high and the design value of engine mixture ratio is less than the peak specific impulse mixture ratio value, this cycle can provide high performance with minimum complexity. If the hydrogen temperature is low and the required turbine flowrates

force the thrust chamber mixture ratio further away from the peak specific impulse value, engine performance can be lower than the alternate cycles. A cycle schematic is shown in Fig. 3 for a single turbine arrangement, and in Fig. 4 for a parallel turbine arrangement.

Auxiliary Heat Exchanger

The auxiliary heat exchanger cycle provides a method to overcome the shortcomings of the hydrogen bleed cycle. They are similar except that the auxiliary heat exchanger uses a separate cooling circuit (possibly a portion of the nozzle) to heat the hydrogen destined for the turbines to a higher temperature than would result from uniform cooling of the entire thrust chamber with all the hydrogen. The maximum temperature attainable is limited by the maximum acceptable heat exchanger wall temperature and the maximum acceptable turbine inlet temperature. The added flexibility of varying the turbine inlet temperature makes it possible to minimize the thrust chamber mixture ratio shift and to adjust the required turbine flow closer to the optimum nozzle base flow. For the cycle selection study, two turbine inlet temperatures were examined, 1500 and 1200 F. A cycle schematic is shown in Fig. 3 for a single turbine arrangement, and in Fig. 4 for a parallel turbine arrangement.

THRUST CHAMBER DESIGN

All cycles use the same thrust chamber design, and all cycles except the auxiliary heat exchanger use the same cooling circuit.

Thrust chamber physical parameters are shown in Table 2.

TABLE 2. SINGLE PANEL THRUST CHAMBER PHYSICAL PARAMETERS

Chamber Length, inches	3
Chamber Contraction Ratio	4
Nozzle Area Ratio	110
Nozzle Percent Length	20

The cooling circuit is shown in Fig. 5.

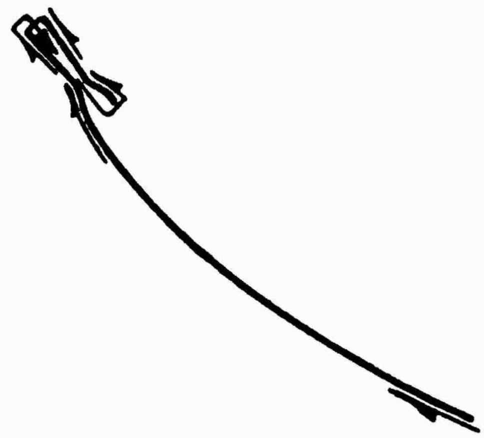


Figure 5. Nozzle First Series Circuit

TURBOMACHINERY ANALYSIS

Turbomachinery

A summary of the ground rules and design limits used for turbomachinery analysis is shown in Table 3 . These limits are within current capabilities and are valid for a relative comparison of the candidate turbine drive cycle.

TABLE 3. TURBOPUMP GROUNDRULES AND DESIGN LIMITS
FOR CYCLE SELECTION COMPARISONS

Maximum Fuel Pump Bearing DN	1.7×10^6
Maximum Oxidizer Pump Inducer Tip to Impeller Tip Diameter Ratio	0.8
Pump Inlet Pressure (Both fuel and oxidizer), psia	30
Maximum Turbine Inlet Temperature, F	1500
Maximum Turbine Tip Speed, ft/sec	1500
Minimum Turbine Blade Height, inches	0.15
Turbine Minimum Mean Diameter, inches	2.00
Turbine Maximum AAN^2	26×10^9

Pumps. A set of preliminary pump characteristics was prepared for the following conditions:

	LO_2	LH_2
Flowrate, lb/sec	46.2	8.4
Stages	1	2

The resulting design point pump conditions for each of the cycles are shown in Tables 4 and 5 .

Off-design pump performance for both pumps was estimated from a nondimensionalized pump map which is typical for the pump design configurations. Because pump off-design performance can be estimated by similarity relations, the map, expressed in terms of head, flow, speed, and efficiency normalized to design values, should provide relatively accurate performance estimates for either the fuel or oxidizer pump.

Turbines. Turbine performance was estimated with gas properties determined from the thrust chamber heat transfer data supplied for the hydrogen drive cycles and an assumed maximum turbine inlet temperature of 1200 and 1500 F for those cycles using combustion gases. These included hydrogen and oxygen pumps driven by parallel turbines, each directly coupled to its pump, and a single turbine operating at the hydrogen pump speed with a gear-driven, reduced-speed oxygen pump. Resulting design point performance is shown in Tables 4 and 5 .

For this preliminary screening, off-design turbine performance was determined by assuming that the turbine pressure ratio was held constant so that, as the pump speed changed at various off-design conditions, the turbine isentropic velocity ratio (u/c), and consequently efficiency, changed proportionally. With detailed turbine and pump performance information and a computerized nonlinear engine model, the turbine pressure ratio can vary as the engine is throttled, resulting in small variations in turbine efficiency (a percentage point or less).

TABLE 4. DRIVE CYCLE TURBOMACHINERY PARAMETER COMPARISON--PARALLEL TURBINES

Design or Operating Parameter	Expander	Staged Topping Combustion	Gas Generator	T/C Tapoff	Hydrogen Bleed	Auxiliary Heat Exchanger	
Engine Mixture Ratio, o/f	5.5	5.5	5.5	5.5	5.5	5.5	5.5
Thrust Chamber Mixture Ratio, o/f	5.57	5.54	5.92	5.92	6.60	6.08	6.16
Secondary Mixture Ratio, o/f	-0-	0.65	0.65	0.65	-0-	-0-	-0-
Secondary Flow Ratio, $\dot{W}_s/\dot{W}_p$	0.002	0.002	0.0209	0.0209	0.0262	0.0149	0.0168
Pump Discharge Pressure, psia							
Fuel (2-stage)	2291	2689	1740	1740	1722	1730	1730
Oxidizer (1-stage)	1174	2689	1221	1221	1174	1174	1174
Turbine Drive Gas Inlet Temperature, °F	380	1500	1500	1500	380	1500	1200
Turbine Inlet Pressure, psia							
Fuel	1619	1375	600	600	600	600	600
Oxidizer	1619	1375	390	390	390	360	360
Turbine Pressure Ratio							
Fuel	1.5	1.25	20	20	20	20	20
Oxidizer	1.5	1.25	13	13	13	12	12
Turbine & Pump Speed, RPM							
Fuel	68770	62175	78200	78200	78200	78200	78200
Oxidizer	24270	51345	24270	24270	24270	24270	24270

TABLE 5. DRIVE CYCLE TURBOMACHINERY PARAMETER COMPARISON--SINGLE TURBINE

Design or Operating Parameter	Expander	Staged Topping Combustion	Gas Generator	T/C Tapoff	Hydrogen Bleed	Auxiliary Heat Exchanger	
Engine Mixture Ratio, o/f	5.5	5.5	5.5	5.5	5.5	5.5	5.5
Thrust Chamber Mixture Ratio, o/f	5.57	5.54	5.83	5.83	6.36	5.94	6.01
Secondary Mixture Ratio, o/f	-0-	0.65	0.65	0.65	-0-	-0-	-0-
Secondary Flow Ratio, $\dot{W}_s/\dot{W}_p$	0.002	0.002	0.0165	0.0165	0.0213	0.0116	0.0132
Pump Discharge Pressure, psia							
Fuel	2045	2608	1740	1740	1722	1730	1730
Oxidizer	1354	2608	1221	1221	1174	1174	1174
Pump Speed, RPM							
Fuel	72540	63585	78200	78200	78200	78200	78200
Oxidizer	24270	49920	24270	24270	24270	24270	24270
Turbine Drive Gas Inlet Temperature, °F	380	1500	1500	1500	380	1500	1200
Turbine Inlet Pressure, psia	1403	1320	600	600	600	600	600
Turbine Pressure Ratio	1.3	1.2	20	20	20	20	20

A preliminary screening of the six candidate cycles and two turbine arrangements was conducted. Based on this comparison, two cycles were selected for further, more detailed evaluations and a final selection.

PRELIMINARY SCREENING

PERFORMANCE AND OPERATING PARAMETER COMPARISON

Vacuum delivered specific impulse versus engine mixture ratio at a chamber pressure of 750 psia and a thrust of 25,000 pounds is shown in Fig. 6 . Performance is shown for all cycles considered and for both single and parallel turbine arrangements (closed cycle performance is independent of turbine arrangement). The figure clearly shows that for each open cycle, the single turbine delivers a significant increase in performance when compared to a parallel turbine arrangement for that cycle. The performance increase is due to more efficient generation of required total pump power with a consequent reduction in secondary flow. The lower secondary flow results in lower thrust chamber mixture ratios for a given engine mixture ratio, especially for those cycles using H_2 as the turbine drive gas. The peak theoretical specific impulse occurs at an engine mixture ratio of approximately 4.5:1. Therefore, increases in secondary flow cause thrust chamber specific impulse and kinetic efficiency to decrease.

For the closed cycles, no change in nozzle base secondary flow occurs when either a single turbine or parallel turbines are used since only the amount of secondary flow required for optimum base performance is used in either case. The closed cycles outperform the open cycles for the same reason the open-cycle single turbine outperforms the open-cycle parallel turbine--less secondary flow at any given point. Figure 6 also shows that both closed cycles have the same performance independent of turbine arrangement. However, they differ considerably in complexity and turbomachinery requirements as discussed in the next section.

Engine performance versus engine mixture ratio for an operating chamber pressure of 150 psia is shown in Fig. 7 . Throttled performance exhibits much the same pattern as mainstage performance except that the scale of differences is decreased.

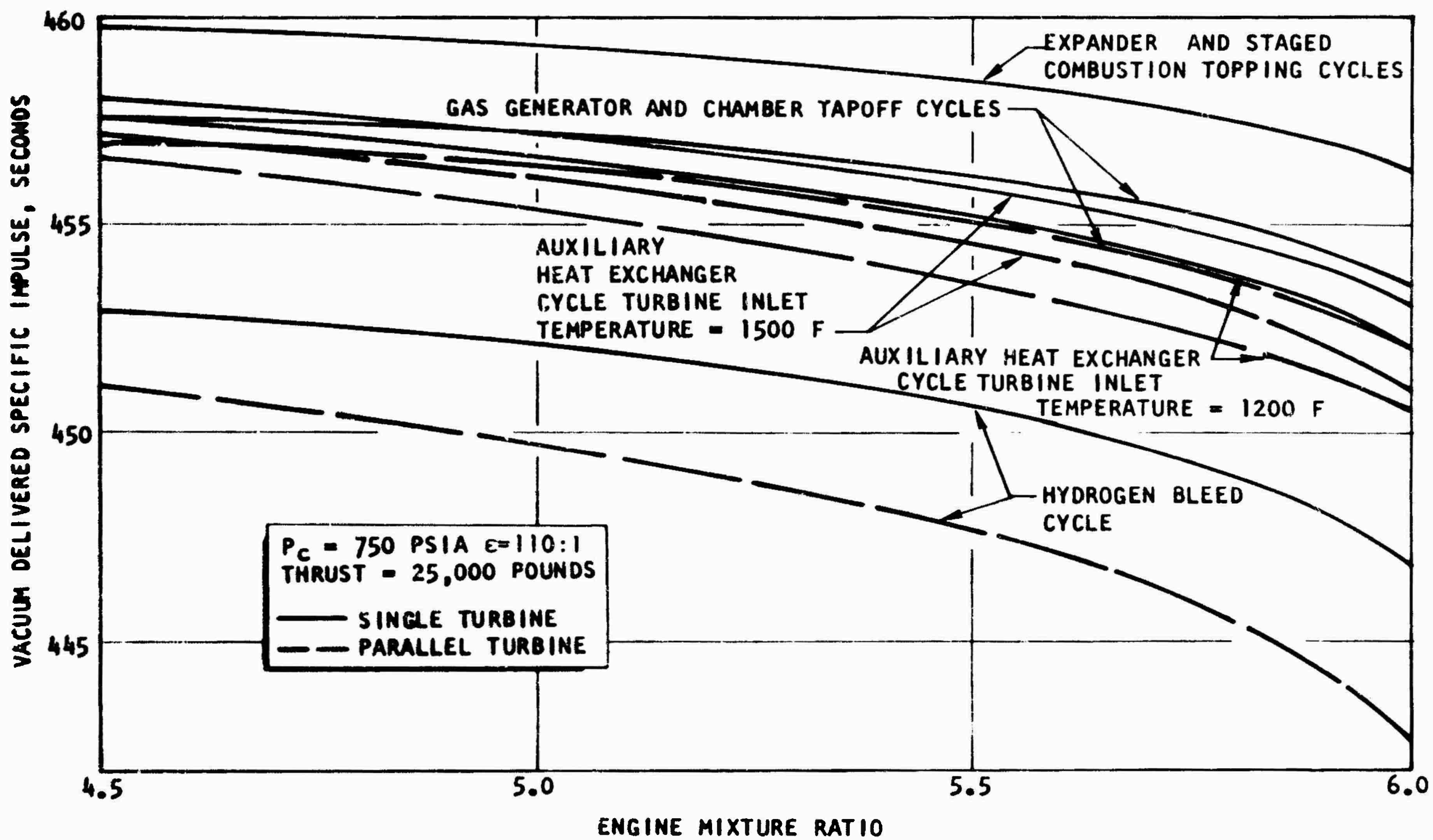


Figure 6. Effect of Turbine Drive Cycle on Aerospike Engine Performance at Design Thrust

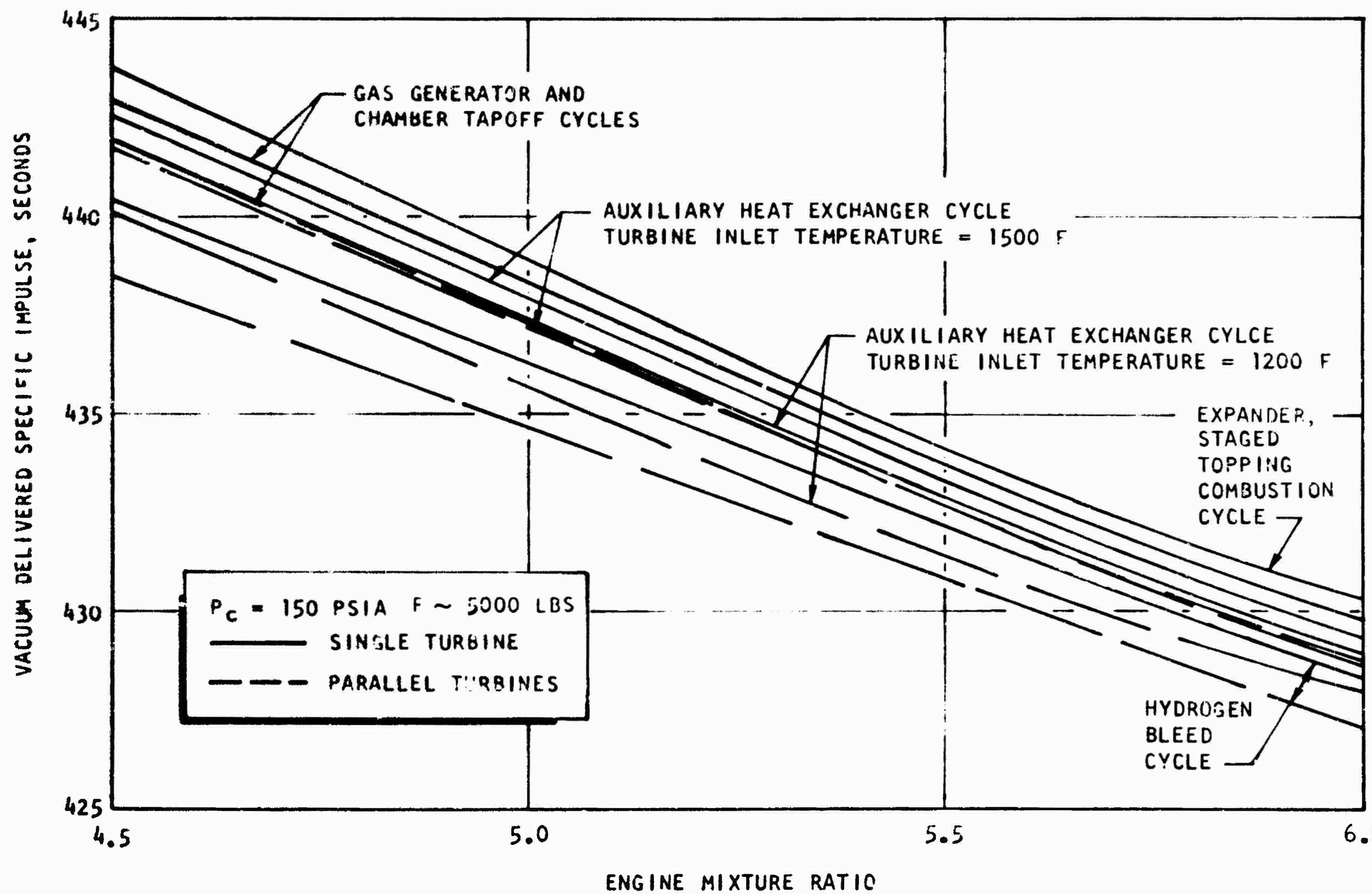


Figure 7. Performance Variation for Various Drive Cycles; Throttled Condition

This lower cycle sensitivity is because the amount of secondary flow required to power the engine, even with the rather inefficient parallel turbine arrangement, is a much smaller percentage of the total flow than was the case at mainstage (i.e., typically 1 percent or less of the thrust chamber flow). Thus, although single turbines and closed cycles still result in lower secondary flow requirements, the effect is not as pronounced as at mainstage. However, single turbines still show higher performance than parallel turbines for the open cycles, and closed cycles still outperform all open cycles. Again, closed cycles show no performance difference between single and parallel turbine arrangements.

OPERATIONAL AND DESIGN COMPLEXITY COMPARISON

The six candidate cycles were evaluated on a relative basis to determine those characteristics that resulted in a noticeable advantage or disadvantage for any specific cycle. The results of this evaluation are summarized in Table 6.

The two closed cycles are considered to be more susceptible to coupling of instabilities through the subsonic flow between the chamber and turbines. The open cycles, having high pressure ratio choked turbines, greatly reduce the possibility of coupled instabilities.

Start and cutoff features of the various cycles can be assessed only on a general basis without at least a preliminary design of the most important engine components and a computerized engine system dynamic model to evaluate start transient behavior of each of the turbine drive cycles. A detailed effort of this type was beyond the study capability. Therefore, it was necessary to depend on past experience and relative comparisons based on specific system features. Those systems that have the higher operating temperature turbine drive gases will probably have higher residual heat in the turbine mass to soak back into the pump and cryogenic propellant feed system components. Heat soak back is an important consideration where immediate or short-time restart capabilities are required as pump childown requirements could increase. In those cycles where the turbine drive

TABLE 6. COMPARISON OF SYSTEM CHARACTERISTICS FOR CANDIDATE DRIVE CYCLES

Operational Feature / Cycle	Expander Topping	Staged Combustion Topping	Hydrogen Bleed	Auxiliary Heat Exchanger	Thrust Chamber Tapoff	Gas Generator
Coupled Feed System Stability	Unchoked turbine in series with thrust chamber is susceptible to coupling	Unchoked turbine in series with thrust chamber is susceptible to coupling				
Start/Cutoff		High turbine heat residual			High turbine heat residual	High turbine heat residual
		Excessive turbine temperature variation possible			Excessive turbine temperature variation possible	Excessive turbine temperature variation possible
	Moderate start time with average repeatability	Moderate start time with average repeatability			Longer start time with poor repeatability	
	Start sensi- tive to initial conditions		Start sensi- tive to initial conditions	Start sensi- tive to initial conditions		

TABLE 6. (Continued)

Operational Feature / Cycle	Expander Topping	Staged Combustion Topping	Hydrogen Bleed	Auxiliary Heat Exchanger	Thrust Chamber Tapoff	Gas Generator
Controlled Variables (in addition to thrust & engine mixture ratio)		Preburner mixture ratio		Heat exchanger flow		GG mixture ratio
Complexity	System sensitivity to component designs	System sensitivity to component designs Additional combustion process Preburner oxidizer gasification Preburner restart with liquid oxidizer			Experimental location of tapoff ports	Additional combustion process GG oxidizer gasification GG restart with liquid oxidizer

TABLE 6. (Concluded)

Structural Feature \ Cycle	Expander Topping	Staged Combustion Topping	Hydrogen Bleed	Auxiliary Heat Exchanger	Thrust Chamber Tapoff	Gas Generator
Materials		High turbine temperature			High turbine temperature	High turbine temperature
Packaging		Preburner				GG
		Preburner oxidizer gasifier				GG oxidizer gasifier
	Thrust chamber manifold for return of turbine drive gas	Thrust chamber manifold for return of turbine drive gas				

gas requires products of closely regulated, low mixture ratio combustion gases, such as in a gas generator, precombustor or thrust chamber tapoff, the possibility of excessive gas temperature variations at the turbine inlet can result in development or operational problems or may require additional closed-loop control capability. Start transient times are strongly influenced by the method by which the energy is added to the turbine drive gas initially during the start transient. Gaseous hydrogen drive cycles that absorb the residual heat from the cooling jacket, for the initial energy to start rotation of the fuel turbopump, will have a relatively faster start time than those relying on complete priming of the fuel side and ignition before generating hot gas for the turbine drive. However, those systems depending on residual heat for the initial starting energy will be more sensitive to initial hardware temperature conditions. Those systems having a preburner or gas generator will require some means of gasifying the liquid oxidizer to achieve the required throttling of the preburner or gas generator.

Certain cycles will result in a more complex engine control system in that they add additional control variables such as preburner or gas generator mixture ratio or auxiliary heat exchanger flowrate versus thrust. Engine system complexity also is considered to increase when components such as the pump, turbine, and thrust chamber are placed in series and their design and operating conditions become interdependent. The addition of a combustion process such as in a gas generator or precombustor results in increased complexity, especially when oxidizer gasification is required for ignition or throttling of this additional combustion chamber. Higher turbine drive gas and operating temperatures limit the materials and design configurations that are feasible for turbines. Additional components required in certain candidate cycles also may present engine packaging problems or integration difficulties such as additional manifolding to collect or return turbine drive gases from or to the individual thrust chamber segments.

CYCLE SELECTION

As a result of this preliminary screening, two candidate drive cycles (one closed cycle and one open cycle) were selected for a more detailed evaluation. The expander topping cycle was selected because it provided the highest delivered engine specific impulse (458.4 seconds), was the least complex of the "closed" cycles of equal performance, and had the lowest turbine-operating temperature. The gas generator cycle also was chosen because it provided the highest specific impulse (456.2 seconds with a single turbine, 455.0 seconds with parallel turbines) of the open cycles. The gas generator cycle also offered fast and repeatable start characteristics with less complex chamber design and development.

These two cycles were compared with both a single turbine (gear-driven oxidizer pump) and parallel turbines, making a total of four systems evaluated. Criteria used in this final comparison included:

1. Component design and operating conditions
2. Engine system weight effects
3. Engine life-influencing factors
4. Engine control system influence
5. Turbopump design features
6. Engine start transient
7. Available power margin
8. Production and development costs

Preliminary engine system power balances were prepared for each of the systems to define required component operating conditions, as shown in Table 7 . The performance of the expander cycle was at least 2.2 seconds higher than the gas generator cycle and was independent of the turbine arrangement where the gas generator cycle performance was affected by turbine arrangement or efficiency

TABLE 7. ENGINE SYSTEM PERFORMANCE COMPARISON FOR SELECTED DRIVE CYCLES

Operating or Design Parameter	Expander Cycle		Gas Generator Cycle	
	Single Turbine	Parallel Turbine	Single Turbine	Parallel Turbine
THRUST (pounds)	25,000	25,000	25,000	25,000
CHAMBER PRESSURE (psia)	750	750	750	750
EXPANSION AREA RATIO	110	110	110	110
ENGINE MIXTURE RATIO	5.5:1	5.5:1	5.5:1	5.5:1
THRUST CHAMBER MIXTURE RATIO	5.57:1	5.57:1	5.83:1	5.92:1
SPECIFIC IMPULSE (seconds)	458.4	458.4	456.2	455.0
SECONDARY FLOW RATIO (τ)	.002	.002	.0165	.021
SECONDARY FLOW MIXTURE RATIO	0	0	0.65:1	0.65:1
HYDROGEN INJECTION TEMPERATURE °F	380	380	380	380
TURBINE INLET TEMPERATURE °F	380	380	1500	1500
PERCENT TURBINE BYPASS	13	11	0	0
TURBINE INLET PRESSURE (psia)	1546	1784/1712	600	600/390
TURBINE PRESSURE RATIO	1.3	1.5/1.44	20	20/13
FUEL PUMP DISCHARGE PRESSURE, PSIA	2238	2476	1796	1796
OXIDIZER PUMP DISCHARGE PRESSURE, PSIA	1354	1204	1257	1207

characteristics. This was explained in the previous discussion of the overall performance comparison. Other important system features were the lower turbine operating temperature and the higher pump discharge pressure associated with the expander cycle.

A weight comparison of those engine system components not common or identical in each of the systems is presented in Table 8 . A maximum weight difference of 36 pounds resulted for these four systems. Using the contract-specified specific impulse/weight exchange factor of 42.6 pounds equivalent to 1.0 second of specific impulse, this weight difference was equivalent to 0.8 second. These weight differences were relatively small and were of minor importance in the overall comparison. However, they were included in the final comparison of overall engine performance.

A control system selection study was conducted for the four engine systems to establish the most favorable method and evaluate control system considerations which could influence the cycle selection. A more-detailed discussion of this study is presented in Appendix A. A summary of the selected control methods and necessary components is presented in Table 9 . The gas generator cycle had the additional control variable gas generator mixture ratio, thus requiring additional control system components.

A summary comparison of the turbopump design features is presented in Tables 10 and 11. The gas generator parallel turbine arrangement provided the lightest turbopump assembly weight, but the low-flow high-pressure-ratio turbines must be partial admission to achieve the minimum blade height. The gear box used in conjunction with the single turbine resulted in two additional bearings and seal packages along with additional rotating components, such as gears and shafts. The low-temperature turbine gas associated with the expander cycle reduced the design criticality of the turbine and related components.

Estimates of engine start transient times are presented in Table 12, illustrating the effect of the four turbopump drive systems, as well as tank pressure and

TABLE 8. COMPARISON OF ENGINE SYSTEM COMPONENT* WEIGHTS FOR CANDIDATE TURBINE DRIVE CYCLES

	Weight, Pounds			
	Expander Topping Cycle		Gas Generator Cycle	
	Single Turbine	Parallel Turbine	Single Turbine	Parallel Turbine
PROPELLANT DUCTING & TURBINE DUCTING	12.5	13.0	4.5	3.3
HOT GAS VALVE	4.0 (2 req.)	6.0 (3 req.)	-	2.0
GAS GENERATOR VALVE	-	-	3.0 (2 req.)	3.0 (2 req.)
OXID. THROTTLE VALVE	3.0		3.0	3.0
OXID. HEAT EXCHANGER	-	-	2.5	2.5
GAS GENERATOR	-	-	6.0	6.0
FUEL TURBOPUMP	124.3	61.0	107.8	47.2
OXID. TURBOPUMP		39.5		40.5
TOTAL	143.8	119.5	126.8	107.5
▲ WEIGHT	+36.3	+12.0	+19.3	Ref.
EQUIV. ▲I_s,sec	- 0.8	- 0.3	- 0.4	Ref.

*Only components not common to each system.

TABLE 9. CONTROL SYSTEM COMPARISON

CYCLE	EXPANDER TOPPING	EXPANDER TOPPING	GAS GENERATOR	GAS GENERATOR
TURBOPUMP CONFIGURATION	GEARED	PARALLEL	GEARED	PARALLEL
CONTROL LOCATIONS				
THRUST	TURBINE BYPASS	TURBINE BYPASS	GG H_2 INLET	GG H_2 INLET
ENGINE MR	O_2 MAIN LINE	O_2 TURBINE INLET	TC O_2 INLET	O_2 TURBINE INLET
GG MR			GG O_2 INLET	GG O_2 INLET
BASE FLOW	BASE FLOW LINE*	BASE FLOW LINE*		
ENGINE SYSTEM ISOLATION	PUMP INLETS	PUMP INLETS	PUMP INLETS	PUMP INLETS
START CONTROL				TC O_2 INLET
NUMBER OF CONTROLLERS				
LIQUID	3	2	3	3
GASEOUS H_2	2	3	1	1
GASEOUS O_2			1	1
HOT GAS				1
CONTROLLERS REQUIRING FEEDBACK	2	2	3	3

* ORIFICE MAY BE SUFFICIENT

TABLE 10. AMPS 25K TURBOPUMP PERFORMANCE AND DESIGN COMPARISON

Criteria 10-Hour Life 300 Cycles	Expander		Gas Generator	
	Single Turbine and Gear	Parallel Turbines	Single Turbine and Gear	Parallel Turbines
TURBINE FLOWRATE, lb/s	7.75	7.36	.912	1.148
TURBINE ADMISSION %, F/O	100	100/100	100	75/50
TURBINE STRESS $\times 10^9$ AAN^2	30.4	28.7	32.8	32.6
BEARING DN $\times 10^6$	1.23	1.17	1.17	1.17
NUMBER OF BEARINGS	6	4	6	4
NUMBER OF SEALS	9	7	9	7
TURBOPUMP WEIGHT, lb	124.3	100.5	107.8	87.7

TABLE 11. TURBOPUMP DESIGN COMPARISON

	Expander Cycle		Gas Generator Cycle	
	Single Turbine	Parallel Turbine	Single Turbine	Parallel Turbine
OPERATING TEMPERATURE	LOW TURBINE GAS TEMP., GEAR BOX PROVIDES THERMAL BARRIER FOR PUMPS, HYDROGEN COOLANT REQUIRED FOR GEARS	LOW TURBINE GAS TEMP.	HIGH TURBINE GAS TEMP., GEAR BOX THERMAL BARRIER, HYDROGEN COOLANT FOR GEARS	HIGH TURBINE GAS TEMP.
BEARING ARRANGEMENT	HIGH RADIAL LOAD ON GEAR BEARINGS, 6 BEARINGS	LOW TEMPERATURE, 4 BEARINGS	HIGH RADIAL LOADS, 6 BEARINGS	HIGH TEMPERATURE SOAK-BACK FROM TURBINE
FABRICATION EASE	23 ROTATING PARTS, GEAR ALIGHMENT	15 ROTATING PARTS, LESS MACHINING, IDENTICAL TURBINES	23 ROTATING PARTS, GEAR ALIGNMENT	15 ROTATING PARTS, LESS MACHINING, IDENTICAL TURBINES

TABLE 12. COMPARISON OF ESTIMATED ENGINE SYSTEM START TIMES FOR CANDIDATE CYCLES

	Expander Topping Cycle		Gas Generator Cycle	
	Single Turbine	Parallel Turbine	Single Turbine	Parallel Turbine
Immediate Restart	3.3 (4.6)	3.0 (4.2)	2.8 (3.6)	2.5 (3.2)
Ambient Start	4.3 (6.0)	3.7 (5.2)	3.8 (4.9)	3.2 (4.1)
Assumes Liquid at Pump Inlet Nominal Tank Pressures = 70 psia ( ) = 35 psia Tank Pressure				

NOTE: Start times to 90 percent thrust, seconds

initial hardware temperature (immediate restart versus restart after long-term coast). The expander cycle had approximately 0.5 second longer engine start times and was somewhat more sensitive to initial hardware thermal conditions and propellant tank pressure. Single turbine systems have greater starting inertia thus requiring longer start times.

Engine system power margin available for development uncertainties was investigated for the expander cycle. This power margin was defined in terms of the additional system pressure drop, reduction in turbine inlet temperature, or reduced turbine efficiencies (that could possibly occur during development) that can be absorbed by the engine system by directing most of the hydrogen bypass flow through the turbines. A minimum bypass of 5 percent of the hydrogen flow was maintained for all conditions to ensure engine control capabilities. The allowable margin in system ΔP was determined for locations between the pump discharge and the turbine inlet, and also for locations downstream of the turbine (increases in system ΔP's downstream of turbine are multiplied by the turbine pressure ratio as they are seen by the pump). The development power margins, along with significant turbopump operating characteristics, are shown in Tables 13 and 14 for the

TABLE 13. EXPANDER CYCLE POWER MARGIN - SINGLE TURBINE [1]

	Hydrogen ΔP Pump Discharge to Turbine Inlet	Hydrogen ΔP Turbine Discharge to P_c	Turbine Inlet Temperature	Turbomachinery Efficiencies[2] Turbine	Oxid. Pump	Fuel Pump	Both Pumps
NOMINAL VALUE	692 psia	1189 psia	840 R	0.730	0.750	0.715	-
ALLOWABLE MARGIN	+1180 psi	+392 psi	-350 R	0.487 (-33.2%)	0.173 (-76.9%)	0.451 (-36.9%)	0.477/ 0.501 (F/O) (-33.2%)
OPERATING CHARACTERISTICS RESULTING FROM ALLOWABLE VALUE OF VARIABLE							
FUEL PUMP SPEED, RPM	63730	71450	80200	80200	80200	80200	[illegible]200
FUEL PUMP DISCHARGE PRESSURE, psia	3775	3222	2595	2595	2595	2595	2595
TURBINE PRESSURE RATIO	1.6	1.6	1.6	1.6	1.6	1.6	1.6
BYPASS FLOW, % OF TOTAL HYDROGEN	5	5	5	5	5	5	5

(1) Pumps and Turbine Are Redesigned at the Allowable Value

(2) At Nominal Velocity Ratios

TABLE 14. EXPANDER CYCLE POWER MARGIN - PARALLEL TURBINES (1)

	Hydrogen ΔP Pump Discharge to Turbine Inlet	Hydrogen ΔP Turbine Discharge to P_c	Turbine Inlet Temperature	Turbomachinery Efficiencies (2)					
				Oxid. Turb.	Fuel Turb.	Both Turb.	Oxid. Pump	Fuel Pump	Both Pumps
NOMINAL VALUE	692 psia	1189 psia	840 R	0.208	0.718	--	0.74	0.715	-
ALLOWABLE MARGIN	+318 psi	+136 psi	-111 R	0.165 (-20.8%)	0.623 (-13.3%)	.655/.190 (F/O) (-8.8%)	0.594 (-20.8%)	0.620 (-13.3%)	0.652/ 0.684 (O/F) (-8.8%)
OPERATING CHARACTERISTICS RESULTING FROM ALLOWABLE VALUE OF VARIABLE									
FUEL PUMP SPEED, RPM	75,850	77,160	80,200	80,200	80,200	80,200	80,200	80,200	80,200
FUEL PUMP DISCHARGE PRESSURE, psia	2,913	2,813	2,595	2,595	2,595	2,595	2,595	2,595	2,595
FUEL TURBINE PRESSURE RATIO	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6	1.6
BYPASS FLOW, % OF TOTAL HYDROGEN	5	5	5	5	5	5	5	5	5

(1) Pumps and Turbines Are Redesigned at the Allowable Value

(2) At Nominal Velocity Ratios

single- and parallel-turbine arrangements. All power margins reflect an assumption that the fuel turbopumps are redesigned at the revised conditions.

The power margin for the expander cycle was found to be adequate to compensate for most unforeseen development problems in the form of reduced efficiencies or increased system ΔP's. The single turbine provided the highest margin because of the lower total turbine flowrate requirement for the nominal single-turbine design. As this available power margin was used up in any of the closed cycles, there was practically no penalty on delivered engine performance. With the gas generator cycle, as additional power requirements were encountered, the increased gas generator flow degraded delivered engine specific impulse. However, theoretically, the power margin was unlimited because the gas generator flow can be increased until all flow passes through the gas generator.

Preliminary engineering estimates of engine system development and production cost were made for the four systems, and are shown in Table 15. The differences in these costs were found to be small and were closely related to weight differences. Those systems using a gear box (single turbines) were less expensive than the parallel-turbine arrangements. The number of control valves required also contributed to some minor differences in the unit production cost. Even though the expander cycle turbomachinery assembly was more expensive, the gas generator and heat exchanger and associated controls tended to equalize the production costs. Specific hardware cost trends were scaled to obtain a total development program cost.

A summary comparison of the selection criteria is shown in Table 16.

As a result of these specific comparisons of the four systems, the expander topping cycle with parallel turbines was selected for the baseline 25,000-pound-thrust aerospike engine system. This selection was based primarily on the maximum performance advantage of this cycle. Other favorable considerations to this selection include the low turbine inlet temperature, least complex turbopumps and controls, and adequate power margin.

TABLE 15. 25K AEROSPIKE ENGINE COMPARISON (PRELIMINARY COST DATA)*

Configuration / Cost Element	Expander Cycle Single Turbine/ Gearbox	Expander Cycle Parallel Turbines	Gas Generator Cycle Parallel Turbines	Gas Generator Cycle Single Turbine/ Gearbox
Difference in First Unit Production Cost (Actual Cost)	-$12K; -1.5% ($808K)	-$7K; -0.9% ($813K)	Baseline ($820K)	-$15K; -1.8% ($805K)
Difference in DDT&E Program Cost (Actual Cost)	-$2.1M; -1.91% ($107.9M)	-$1.4M; -1.27% ($108.6M)	Baseline ($110M)	-$1.4M; -1.27% ($108.6M)

*Final cost data are presented in the Engine Development Program section.

TABLE 16. SUMMARY COMPARISON OF SELECTION CRITERIA

	Expander		Gas Generator	
	Single/Gear	Parallel	Single/Gear	Parallel
Specific Impulse, I_s, seconds	458.4	458.4	456.2	455.0
Δ I_s, seconds	3.4	3.4	1.2	
Weight Equivalent I_s^*, seconds	-0.8	-0.3	-0.4	Ref.
Effective Δ I_s, seconds	2.6	3.1	0.8	Ref.
Δ Payload**, pounds	410	485	125	Ref.
Life	Least Critical		Most Critical	
Controls				
Controllers	5	5	5	6
Feedback Loops	2	2	3	3
Turbopump	Low turbine temperature Gear box	Low turbine temperature	High turbine temperature Gear box	High turbine temperature
Engine Start	Sensitive to inlet and thermal condition	Sensitive to inlet and thermal condition	Less sensitive Faster start	Less sensitive Faster start
Power Margin	Increased, I_s unaffected	Adequate, I_s unaffected	Maximum, I_s degraded	Maximum, I_s degraded
Cost	← Insignificant Difference →			

$$^*\frac{\Delta \text{ Engine Weight}}{\Delta \text{ Specific Impulse}} = 42.6 \text{ lb/sec} \qquad ^{**}\frac{\Delta \text{ Payload}}{\Delta \text{ Specific Impulse}} = 157 \text{ lb/sec}$$

Parametric engine system performance and weight information was generated, as a part of this overall program effort, for a range of thrust level, chamber pressure, area ratio, and mixture ratio values and for both open and closed turbine drive cycles and single-panel and double-panel cooling circuits. An optimization analysis also was conducted, based on this parametric information, to establish the optimum engine design point configurations for selected design thrust levels. The results of this analysis are presented in the next section for the single- and double-panel, 25,000-pound-thrust engine systems. The closed cycles resulted in the highest system performance for the double-panel engine systems. The remaining cycle selection criteria were then reviewed to determine the applicability of this selection study to the double-panel engine system. The conclusions were found to be equally valid for the selected double-panel, 25,000-pound-thrust engine system.

25,000-POUND-THRUST ENGINE SYSTEM DESIGN POINT SELECTION

The design values of chamber pressure and area ratio were specified for the single-panel engine system. However, with the addition of a double-panel engine to the 25,000-pound-thrust system studies, it was necessary to establish the design value of chamber pressure and area ratio for the component and system design and analysis.

The parametric engine performance and weight information that was generated as a part of this program effort was used to conduct an optimization analysis for the 25,000-pound-thrust aerospike engine systems.

For a fixed engine mixture ratio, the payload capability of an engine in a given mission depends on its delivered specific impulse and weight. Because specific impulse can quite often be purchased at the expense of weight and because the exchange factors on specific impulse and weight are seldom the same (e.g., in a high-energy mission such as the low earth orbit-to-synchronous orbit mission, specific impulse is highly favored in relation to weight), it follows that the optimum engine configuration is not necessarily the lowest weight and/or the highest specific impulse configuration.

To facilitate determination of the optimum configuration, the parametric engine weight and performance data presented in the aerospike parametric information report may be replotted as specific impulse versus engine dry weight for a given thrust level, mixture ratio, and cycle. Constant lines of chamber pressure and expansion area ratio also are shown. If cooling limits and power limits are superimposed on this plot, a graph showing the feasible area of operation for the cycle at the given thrust level and engine mixture ratio results. This graph, as shown in Fig. 8, can be used to determine the optimum engine configuration for any mission by plotting straight lines having as their slope the ratio of the mission engine weight exchange factor to the mission specific impulse exchange factor, i.e.,

$$\left\{\frac{\Delta PL}{\Delta W_e}\right\} \bigg/ \left\{\frac{\Delta PL}{\Delta I_s}\right\} = \left\{\frac{\Delta I_s}{\Delta W_e}\right\}$$

Each line represents constant payload for the given mission. Payload is increased by moving upwards and to the left in a direction perpendicular to the straight lines. The point of tangency between the constant payload line and the uppermost and left-most point in the feasible region of operation represents the optimum engine configuration.

The location of this point of tangency is shown in Fig. 8 for both the single-panel and double-panel engine systems. In the parametric engine information studies, the estimated weights of the two systems were found to be equal for the identical values of chamber pressure; therefore, the feasible region of operation of the double-panel engine system also contains that of the single-panel engine system. The only difference exists in the location of the upper boundary to the regions, which is defined by the cooling limit for the specified cooling circuit (single panel or double panel). The single-panel thrust chamber is limited to lower values of area ratio for a given chamber pressure.

The results of this analysis established an optimum double-panel engine design point of 1000-psia chamber pressure and 200:1 nozzle expansion area ratio.

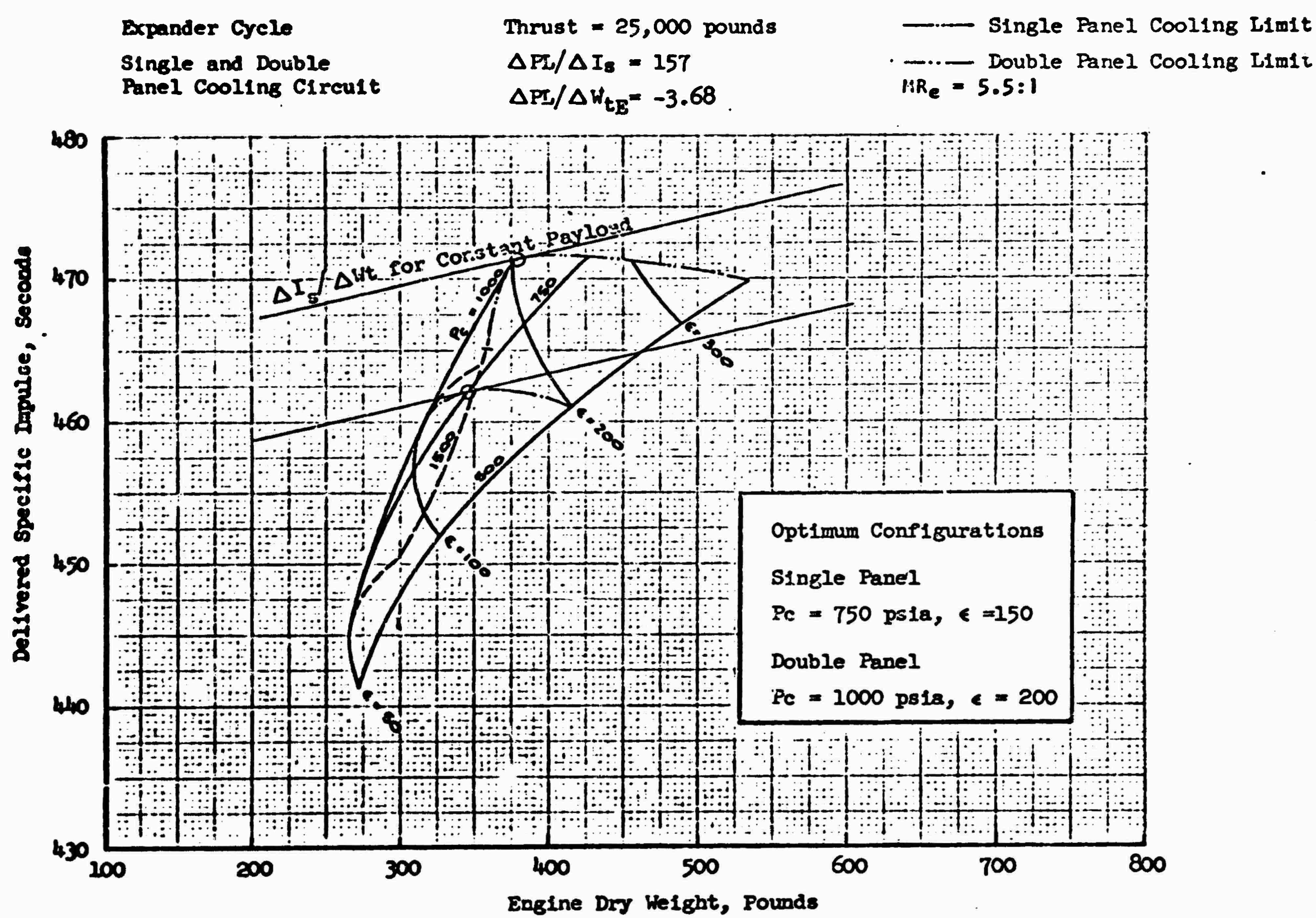


Figure 8. Aerospike Engine Optimization

The optimum single-panel, 25,000-pound-thrust engine system design point does not correspond to the baseline single-panel engine system specified by the contract work statement and described in this report. This occurred in order to reduce the risk of the demonstrator thrust chamber program by providing reduced total heat load to the thrust chamber. Further refinements in the single-panel regenerative coolant system design parameters and circuitry have also contributed to this margin between the selected and optimum engine design points.

DESIGN AND ANALYSIS

The design and analysis section of this report provides a detailed description of the two baseline 25,000-pound-thrust aerospike engine systems: a single-panel thrust chamber cooling circuit (H_2 coolant only) and a double-panel thrust chamber cooling circuit (H_2 and O_2 cooling). As each topic of the engine system design and analysis and component descriptions is discussed, the double-panel engine will be presented first, followed by a description of the single-panel engine. In some cases, the operating features or component designs for the two engines are identical and the same discussion applies to both engines.

Both engine designs have been carried through this preliminary design study phase pending the results of the thrust chamber segment test effort that is being conducted concurrently with, and beyond the term of, this design analysis and under the same contract. A selection of either the single-panel or double-panel thrust chamber cooling concept will be made based on the results of this segment test effort. The detailed design of the two thrust chamber concepts is, therefore, not yet complete and cannot be finalized until this thrust chamber segment test effort is complete. When the final thrust chamber description is available, all engine system flowrates, temperatures, and pressures will be re-evaluated, and the final component operating and design parameters for the selected engine will be established. For this reason, some of the details of the engine system design such as packaging and interconnects are not included in this report.

At quarterly intervals during the aerospike engine system study, an aerospike engine design report was published (Ref. 1), providing a detailed definition and design description of both engine systems and the individual components. As thrust chamber test and design information is acquired, a revised edition of the engine system design report will be published quarterly during the remainder of the thrust chamber development effort. Revised engine balances will be presented based on the then currently available segment design and test information.

ENGINE SYSTEM DESCRIPTION

The baseline single-panel engine system design point corresponds to the demonstrator thrust chamber configuration. The double-panel engine design point was selected based on the results of an engine system optimization study. Both engines are designed for 25,000 pounds thrust at a nominal mixture ratio of 5.5:1. An expander drive cycle is used to provide turbine power. The specific engine configurations were selected based on extensive optimization studies summarized in a later section of this report. Engine operating capability is summarized in Table 17.

The following discussions will present the baseline 25,000-pound-thrust aerospike engine and component descriptions. The optimum double-panel engine system will be presented first, followed by a description of the single-panel engine system.

DOUBLE-PANEL ENGINE SYSTEM DESCRIPTION

The baseline double-panel O_2/H_2 AMPS engine system (shown in Fig. 9) is based on an aerospike thrust chamber design with a maximum thrust level of 25,000 pounds and a throttling capability to 5,000 pounds. The maximum operating chamber pressure is 1000 psia and the aerospike thrust chamber provides an expansion area ratio of 200:1. The nominal operating engine mixture ratio is 5.5:1 with an off-design operational capability of ±0.5 mixture ratio units for propellant utilization purposes. The thrust chamber and nozzle are regeneratively cooled with the fuel flow, and the inner wall of the combustion chamber has a secondary cooling jacket where the oxygen flows in a single up-pass providing a greater overall regenerative cooling capability. Thus, the combustion chamber is fed with heated, gaseous hydrogen and oxygen. The injector incorporates a triplet element configuration.

TABLE 17. AEROSPIKE ENGINE OPERATING CAPABILITY

Propellants	Liquid Oxygen/ Liquid Hydrogen
Nominal Engine Mixture Ratio	5.5:1
Engine Mixture Ratio Operating Range*	Nominal ±0.5
Vacuum Thrust Throttling Capability*	5.0:1
NPSH (O_2/H_2), feet	16/60
Square Pattern Gimbal Angle, degrees	5
Gimbal Acceleration, rad/sec^2	5
Number of Vacuum Starts	
Between Inspections or Servicing	60
Between Overhauls	300
Accumulated Firing Time, hours	
Between Inspections or Servicing	2
Between Overhauls	10
Maximum Single-Run Duration, seconds	1000
Maximum Storage Time in Orbit (Dry), weeks	52
Maximum Time Between Firings (Coast Time), days	14
Minimum Time Between Firings (Coast Time), minutes	10
Ground-Based Maintenance	
Fail-Safe Design	
Autogenous Propellant Tank Pressurization	

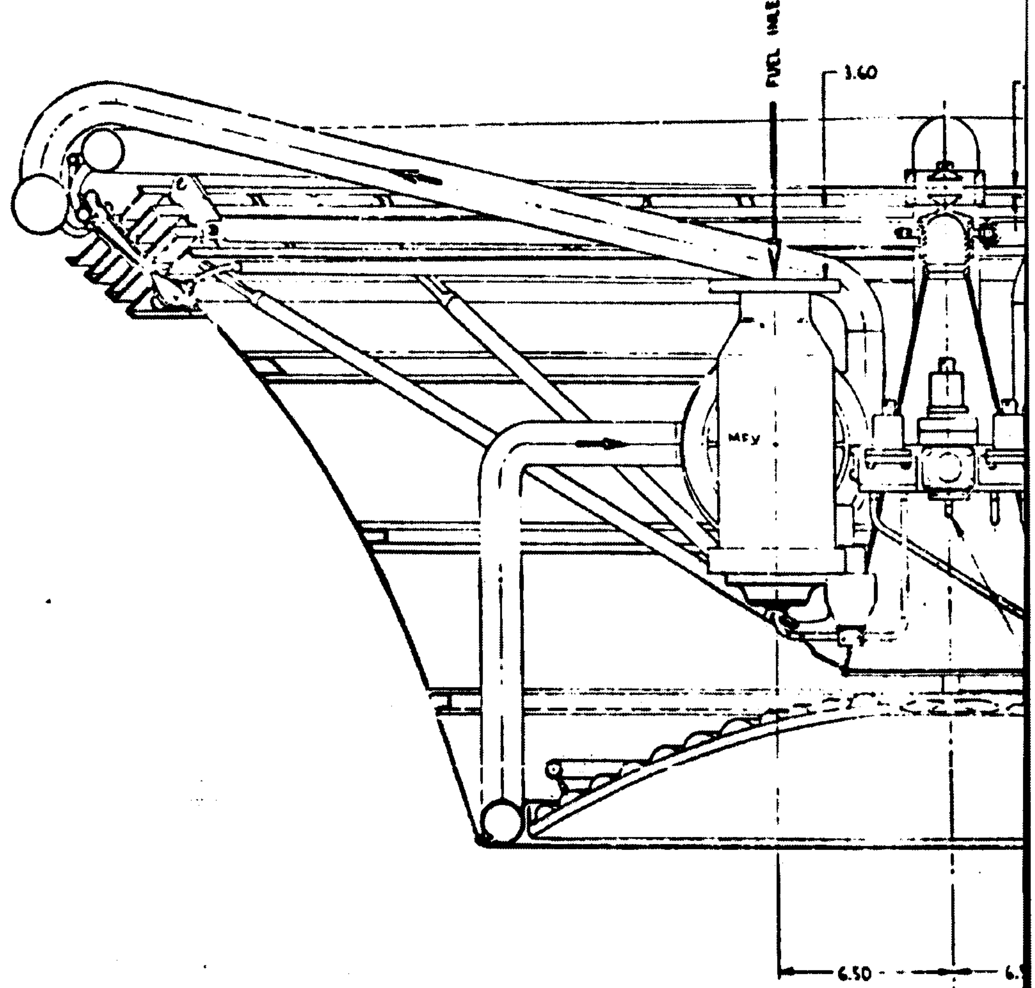
*Continuous

The engine is pump fed by centrifugal-type pumps which are directly driven by axial flow impulse turbines. The expander topping cycle has been selected as a superior turbopump drive cycle for this engine, based primarily on performance and engine system simplicity. The heated hydrogen, after passing through the thrust chamber regenerative cooling circuit, is fed to the two turbines in a parallel flow arrangement. The turbine exhaust flow, except for a small flow which is directed to the base of the aerospike nozzle (approximately 0.2 percent of the total thrust chamber flow) is then directed to the injector fuel inlet manifold.

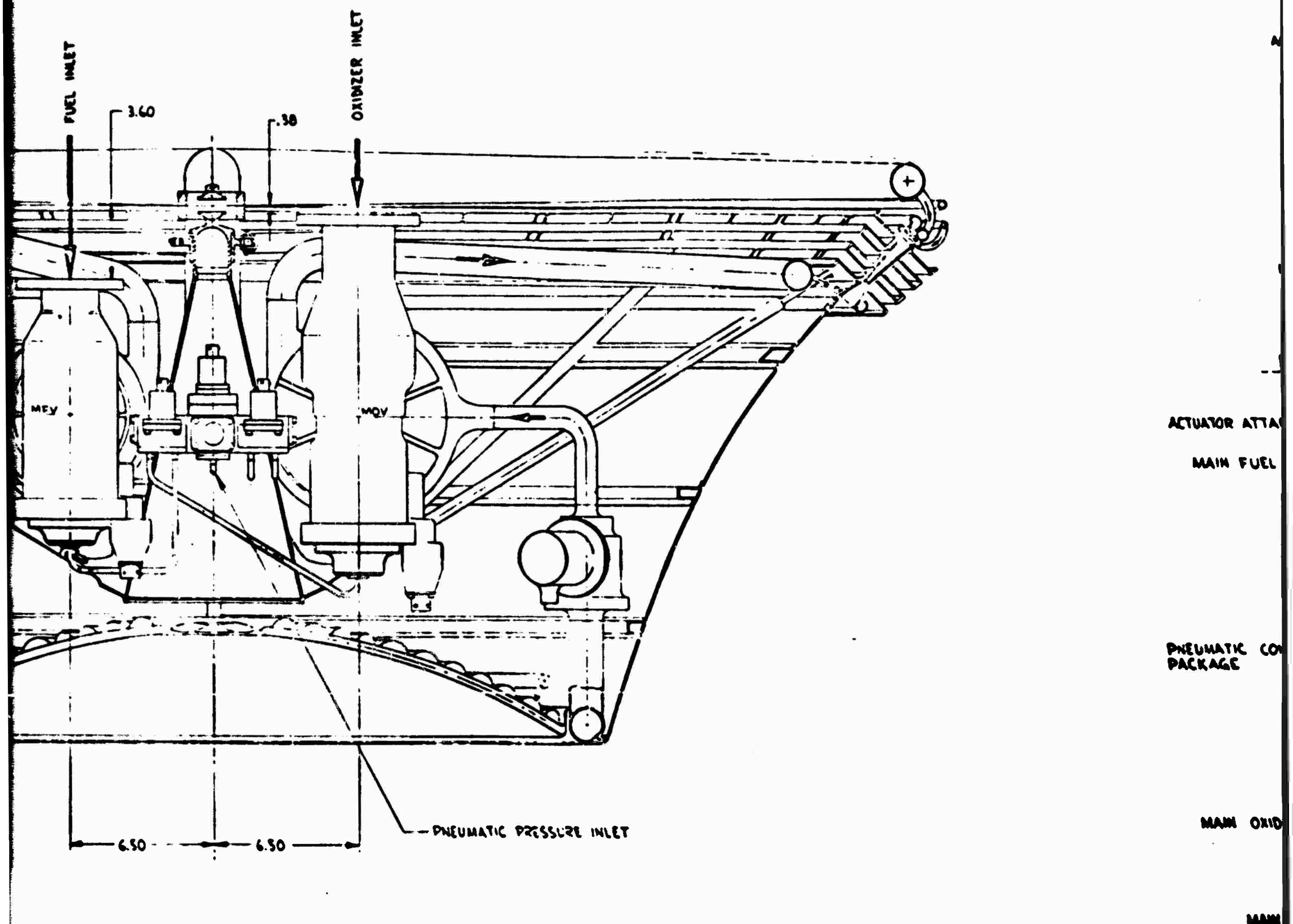
The engine control system consists of main propellant valves (two liquid valves) located upstream of the turbopumps, an oxidizer turbine inlet control valve and a turbine bypass valve (two hot hydrogen valves). The main propellant valves are pneumatically actuated and the variable area turbine control valves are electrically actuated. The turbine bypass valve provides thrust control by varying the amount of heated hydrogen that passes through the turbines. An adequate power and control margin is provided at the nominal full thrust operating point by designing for a nominal bypass flow equal to 24 percent of the total fuel flow. Engine mixture ratio control capability is provided by the oxidizer turbine inlet valve. Engine operation is controlled by a system which receives guidance system commands and engine parameter feedback, and then computes the engine control signals.

The engine is designed for multiple starts at altitude. Engine start is accomplished through a propellant tank-head start sequence. The hydrogen is pressure fed from the main tanks through the thrust chamber regenerative cooling jacket where the hydrogen temperature is increased due to the residual heat capacity of the chamber. The warm hydrogen then passes through the turbines providing the initial power to the fuel and oxidizer pumps according to the prescribed sequence. After ignition, the engine bootstraps itself to full power.

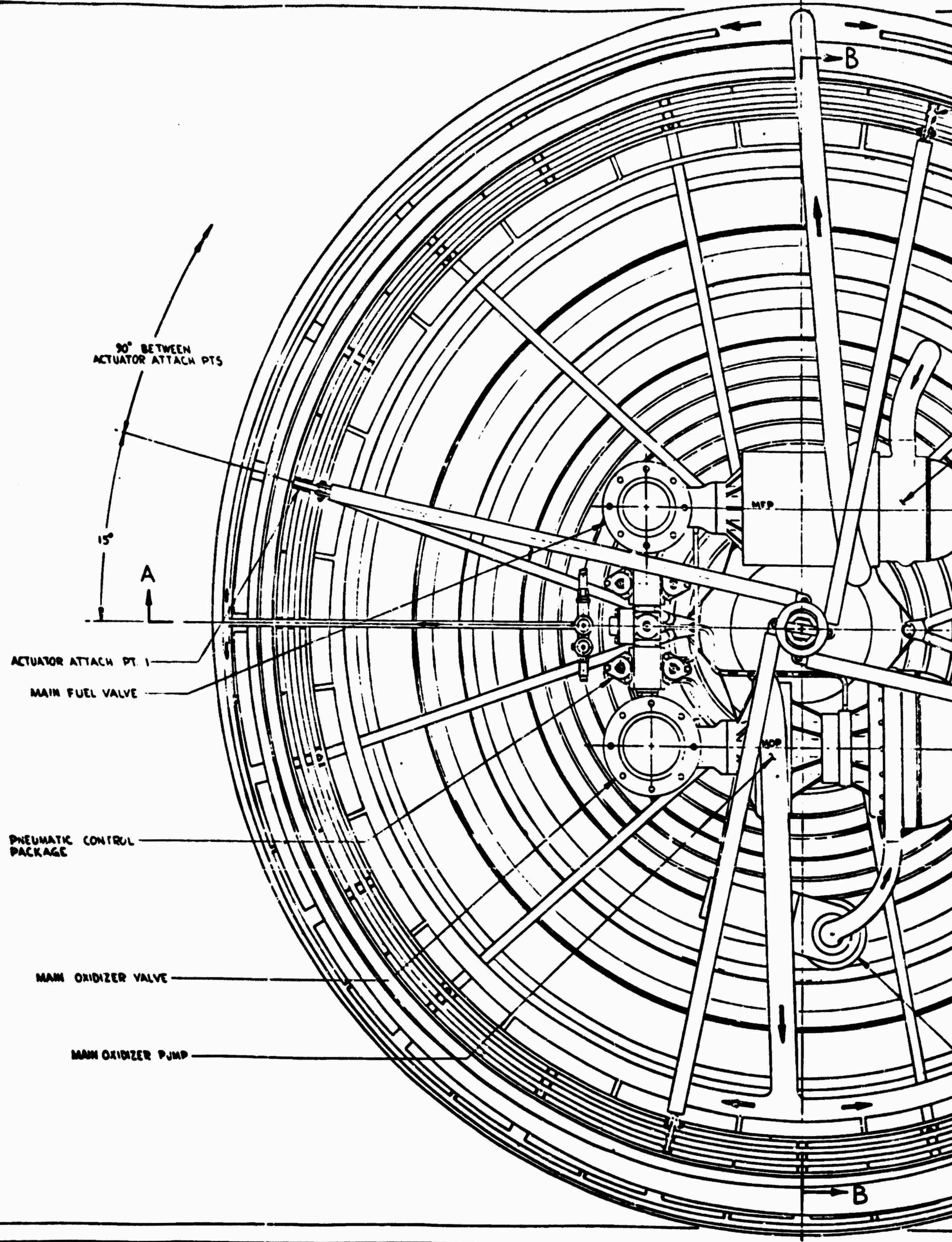
A combustion wave ignition system is used where a spark-induced combustion wave passes through an unburned, gaseous oxygen/hydrogen mixture to ignite a pilot element within each combustion chamber segment. A central mixing chamber/gas generator is used to mix and distribute the propellants through the delivery tubes



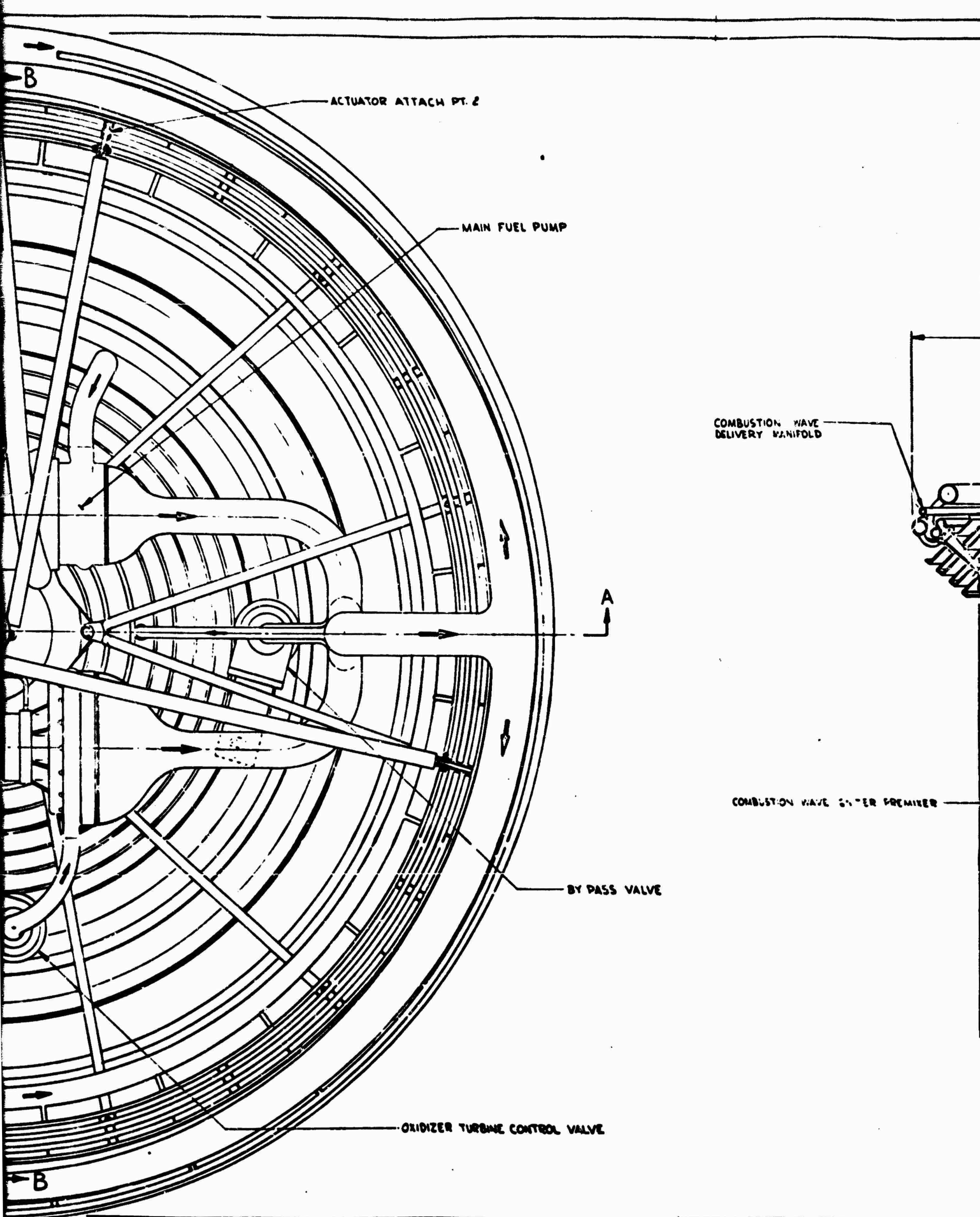
SECTION B-B
(PUMPS & VALVES NOT SEC
(ROTATED 90°)

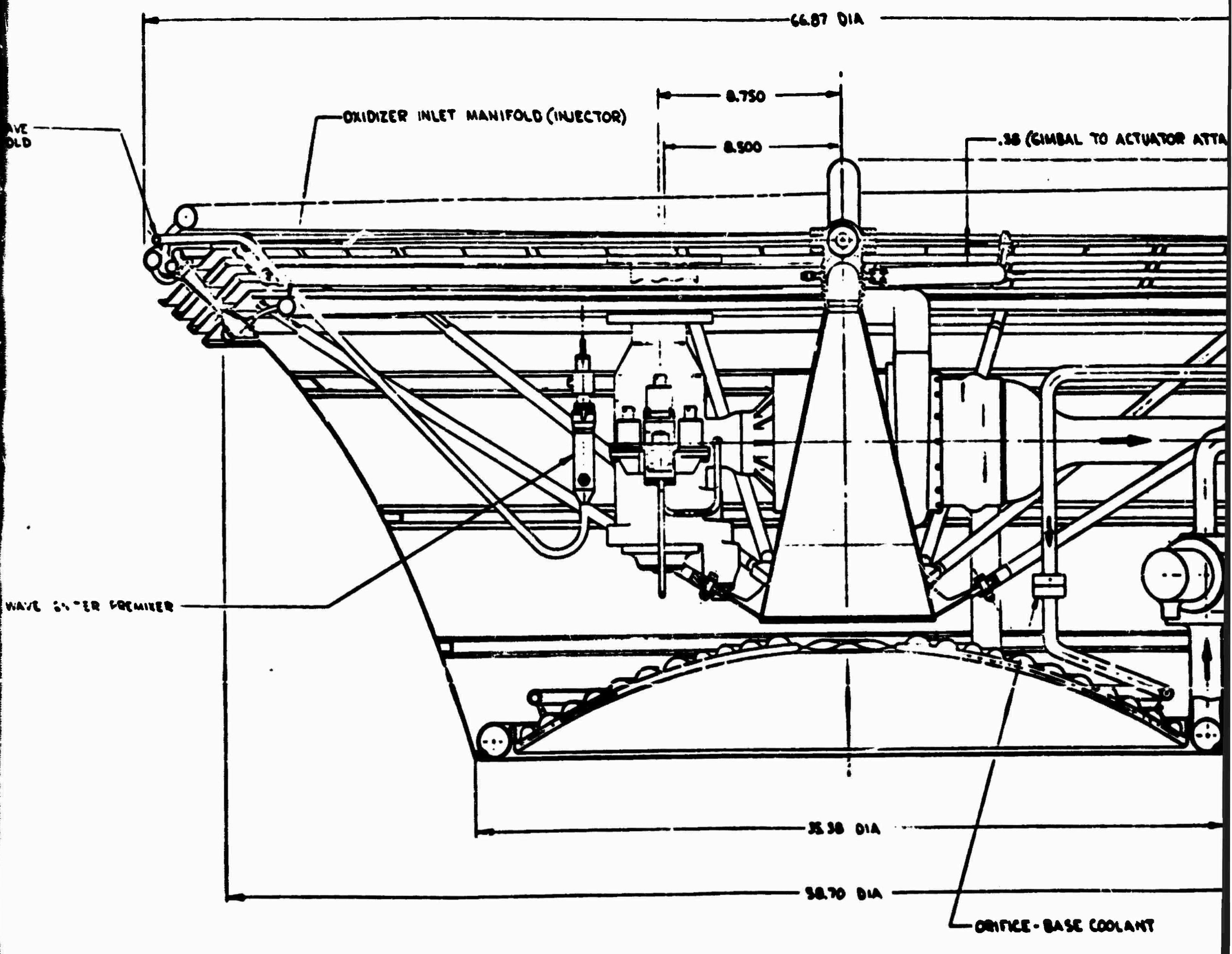


SECTION B-B
(PUMPS & VALVES NOT SECTIONED)
(ROTATED 90°)



90° BETWEEN
ACTUATOR ATTACH PTS
15°
A
B
ACTUATOR ATTACH PT. 1
MAIN FUEL VALVE
PNEUMATIC CONTROL
PACKAGE
MAIN OXIDIZER VALVE
MAIN OXIDIZER PUMP
MFP
MOP
B


ACTUATOR ATTACH PT. 2
MAIN FUEL PUMP
COMBUSTION WAVE
DELIVERY MANIFOLD
A
BY PASS VALVE
OXIDIZER TURBINE CONTROL VALVE
B
B


66.87 DIA
8.750
8.500
OXIDIZER INLET MANIFOLD (INJECTOR)
.38 (GIMBAL TO ACTUATOR ATTA
35.38 DIA
38.70 DIA
ORIFICE - BASE COOLANT

SECTION A-A

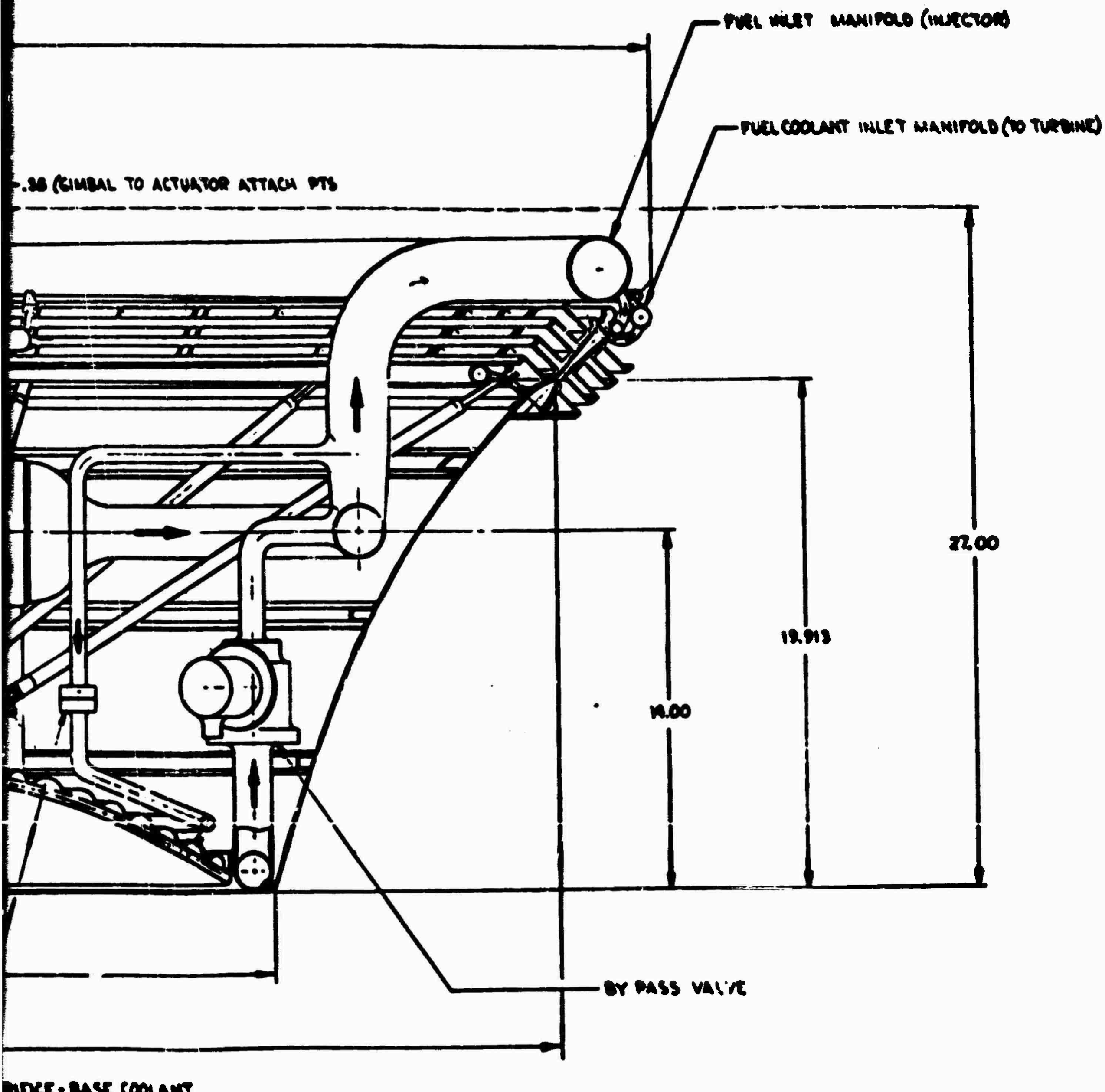


Figure 9. 25,000-Pound-Thrust, Double-Panel Aerospike Engine System

to each segment. The combustion wave is initiated by an electrical arc discharge spark in the premix chamber at the same time that the oxidizer flow to the premix chamber is cut off. The resulting combustion wave then propagates in the unburned mixed propellants toward the segments. Fuel is fed to the premix chamber from the cooling jacket exit and oxidizer is fed from the pump discharge under tank pressure for the initial part of the start transient.

The engine utilizes helium gas for main valve actuation, oxidizer turbopump seal cavity purge, and propellant system purges.

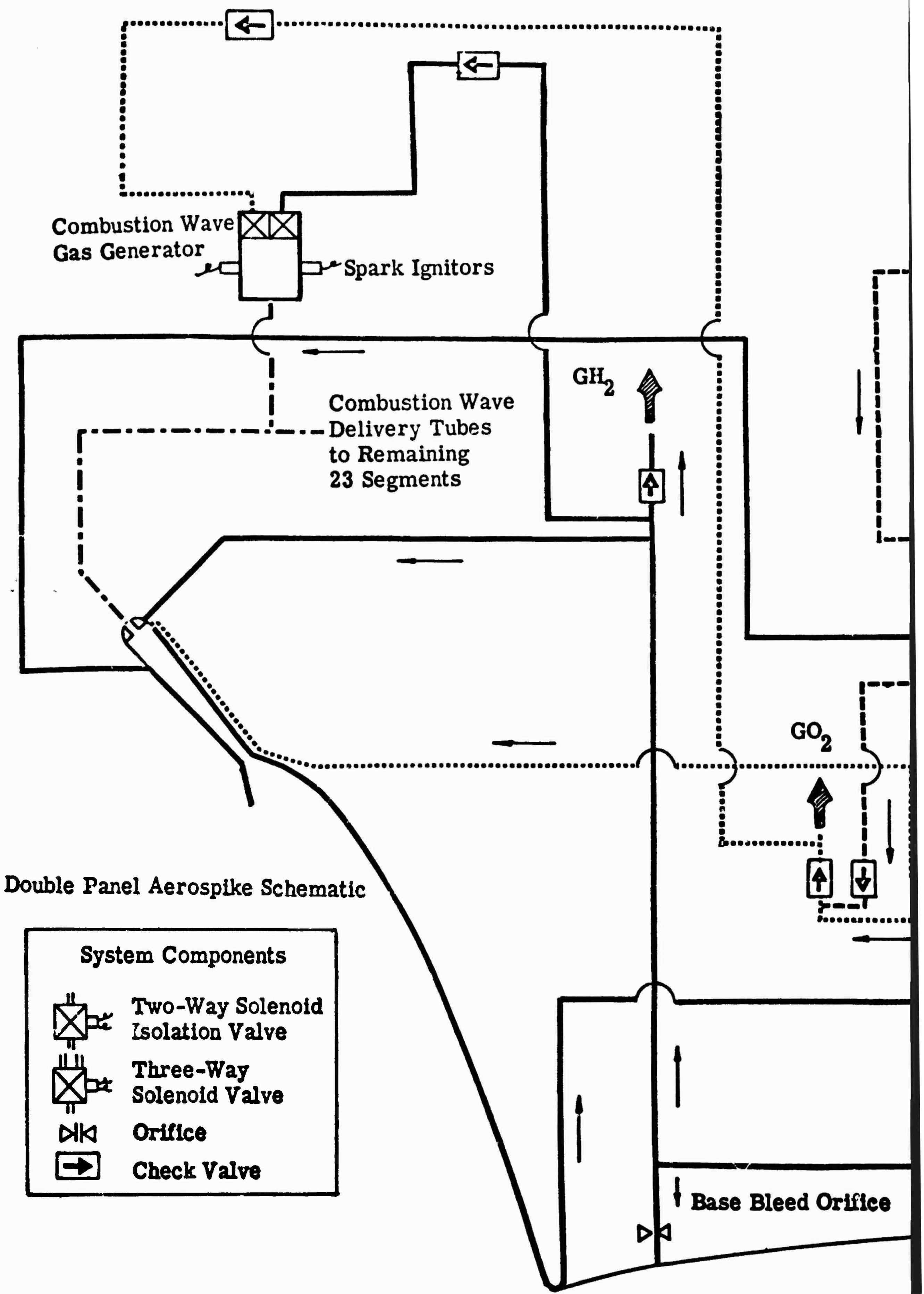
The basic engine system design parameters are shown in Table 18. An engine system propellant and pneumatic flow schematic is shown in Fig. 10. A curve of delivered engine performance over the operating range of thrust and mixture ratio is shown in Fig. 11.

Engine System Weight and Mass Properties

The component weights for the selected 25,000-pound-thrust, double-panel aerospike engine system are shown in Table 19. The double-panel chamber design incorporates a brazed-on copper closeout sheet over the coolant channels in the cast segment outer wall. A double-panel closure is brazed into position on the inner body wall of the segment. The projected thrust chamber weight of 187 pounds is based on 0.010-inch wall thickness and other production-type weight-saving techniques that are considered too costly for the demonstrator thrust chamber. A contingency weight of 19 pounds is included to account for uncertainties in the weight estimates of those components that have not as yet received detailed design attention and possible future production weight influences.

TABLE 18. DOUBLE-PANEL ENGINE SYSTEM DESIGN PARAMETERS

Thrust, pounds		25,000
Chamber Pressure, psia		1000
Expansion Area Ratio		200
Engine Mixture Ratio		5.5
Thrust Chamber Mixture Ratio		5.57
Specific Impulse, seconds		470.4
Base Flowrate, pounds/second		0.10
H_2 Injection Temperature, °R		1097
Turbine Inlet Temperature, °R		1169
Percent Turbine Bypass		24
Turbine Inlet Pressure, psia	Fuel	1870
	Oxidizer	1781
Turbine Pressure Ratio	Fuel	1.57
	Oxidizer	1.31
Pump Discharge Pressure, psia	Fuel	3156
	Oxidizer	1781
Engine Length, inches		27
Engine Diameter, inches		67



Double Panel Aerospike Schematic

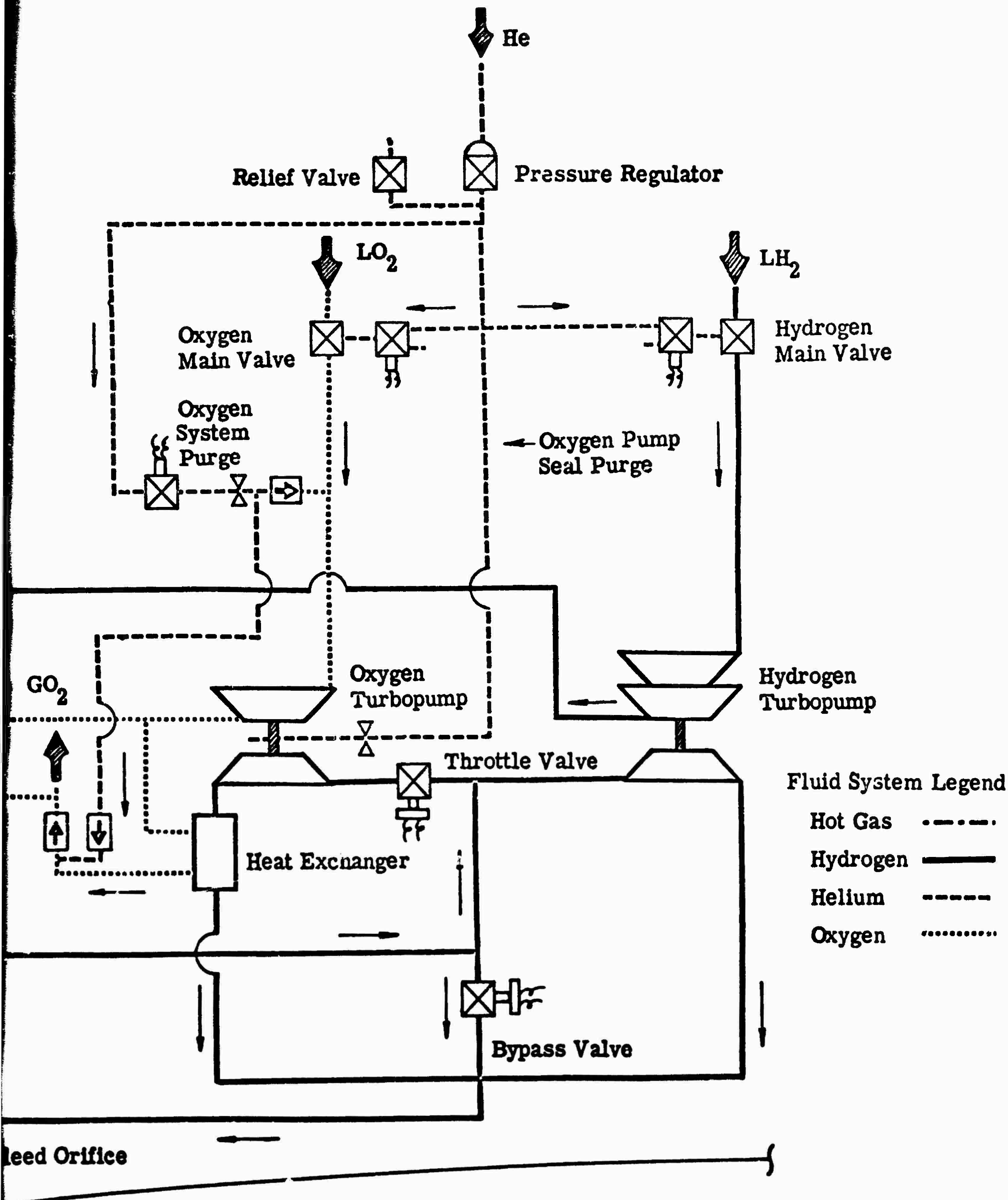


Figure 10. Double-Panel Aerospike Engine Schematic

2

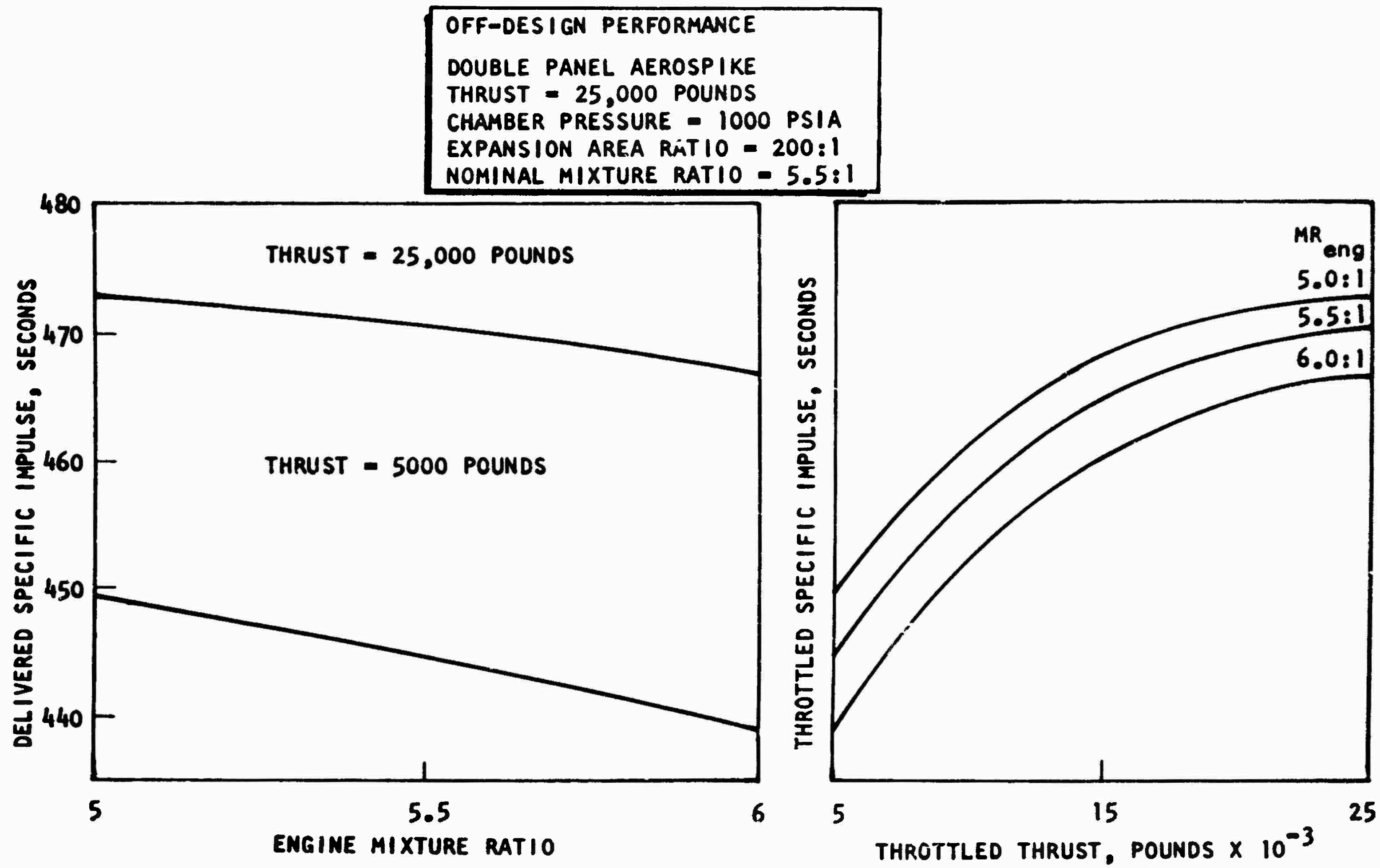


Figure 11. Off-Design Performance

TABLE 19. 25,000-POUND THRUST BASELINE DOUBLE-PANEL AEROSPIKE ENGINE SYSTEM WEIGHTS

Thrust, lbs	25,000
Chamber Pressure, psia	1,000
Expansion Area Ratio	200:1
Nozzle % Length	20

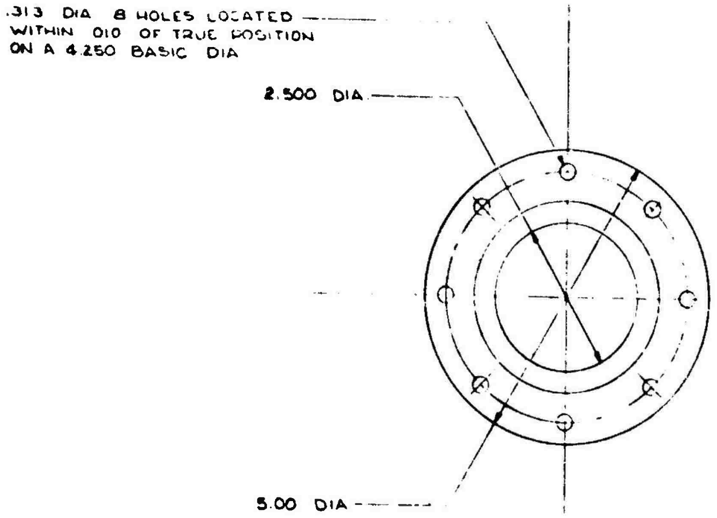
Subsystem Weights	
Combustion Chamber and Shroud	114
Nozzle	45
Base Closure	9
Thrust Mount and Gimbal Assembly	19
Turbopumps and Mounts	85
Propellant Ducting and Inlet Valves	64
Hot-Gas Valves	16
Controls and Miscellaneous	20
Ignition System	7
Contingency	19
Total Engine System Weight, pounds	398

Engine Interface Requirements

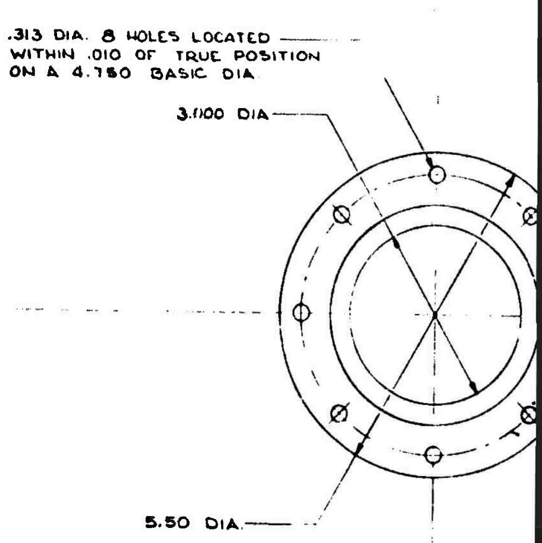
Structural, fluid, and electrical interface connections between the engine system and propellant feed system have been specified. The locations and dimensions of the structural mountings and main propellant inlets to the engine system are shown in Fig. 12. Structural connection for transmission of thrust to the propellant feed system is made at the forward face of the gimbal mount. Gimbal actuator attachment is made at two locations 90 degrees apart at locations where the thrust structure attaches to the thrust chamber. Flexibility in the fuel and oxidizer propellant inlet ducts for engine gimbaling, thermal growth, and manufacturing misalignment is provided on the propellant feed system side of the interface.

Engine System Electrical Requirements. Engine electrical power is required in the following areas: gimbal actuation, engine control valve actuation (oxidizer turbine control valve and turbine bypass valve), solenoid valve actuation, instrumentation, engine controller package, and ignition. The types and numbers of electrical elements are presented in Table 20 along with the preliminary estimates of voltage, current, and power requirements. Total electrical energy storage requirements also are indicated for a typical mission and for a 2-hour-duration, 60-thermal-cycle service life. The total electrical energy storage requirements range from 338 watt-hours for a typical mission to 1674 watt-hours if the full 2.0-hour and 60-start service life is required before battery replacement or servicing. Total energy requirement of the gimbal actuator was estimated somewhat arbitrarily. If more detailed mission thrust vector control information becomes available, this estimate should be re-evaluated as it comprises the major portion of the total electrical energy requirement. Power requirements are included for spark ignition of the premix chamber in the combustion wave ignition system.

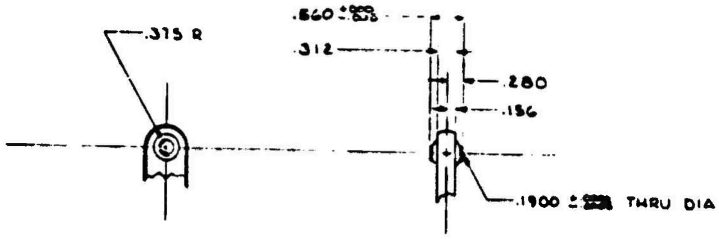
Pneumatic Requirements. The engine system pneumatic package supplied regulated helium for opening the main propellant valves, purge flow to the seal cavity of the oxidizer turbopump, and purge flow to the oxidizer ducts downstream of the main oxidizer valve and the oxidizer heat exchanger. Purge of the fuel side of



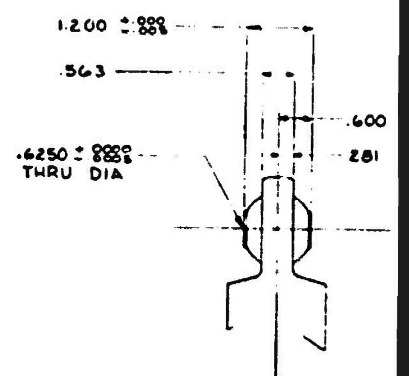
FUEL INLET
(FULL SIZE)



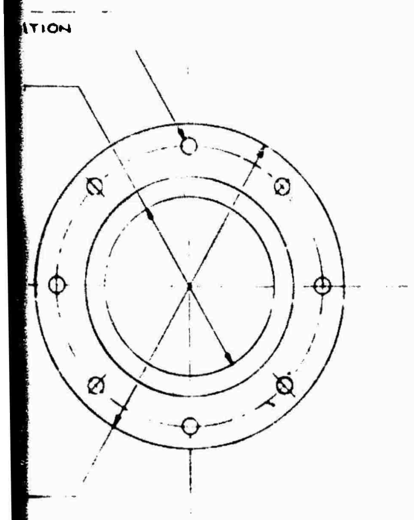
OXIDIZER INLET
(FULL SIZE)



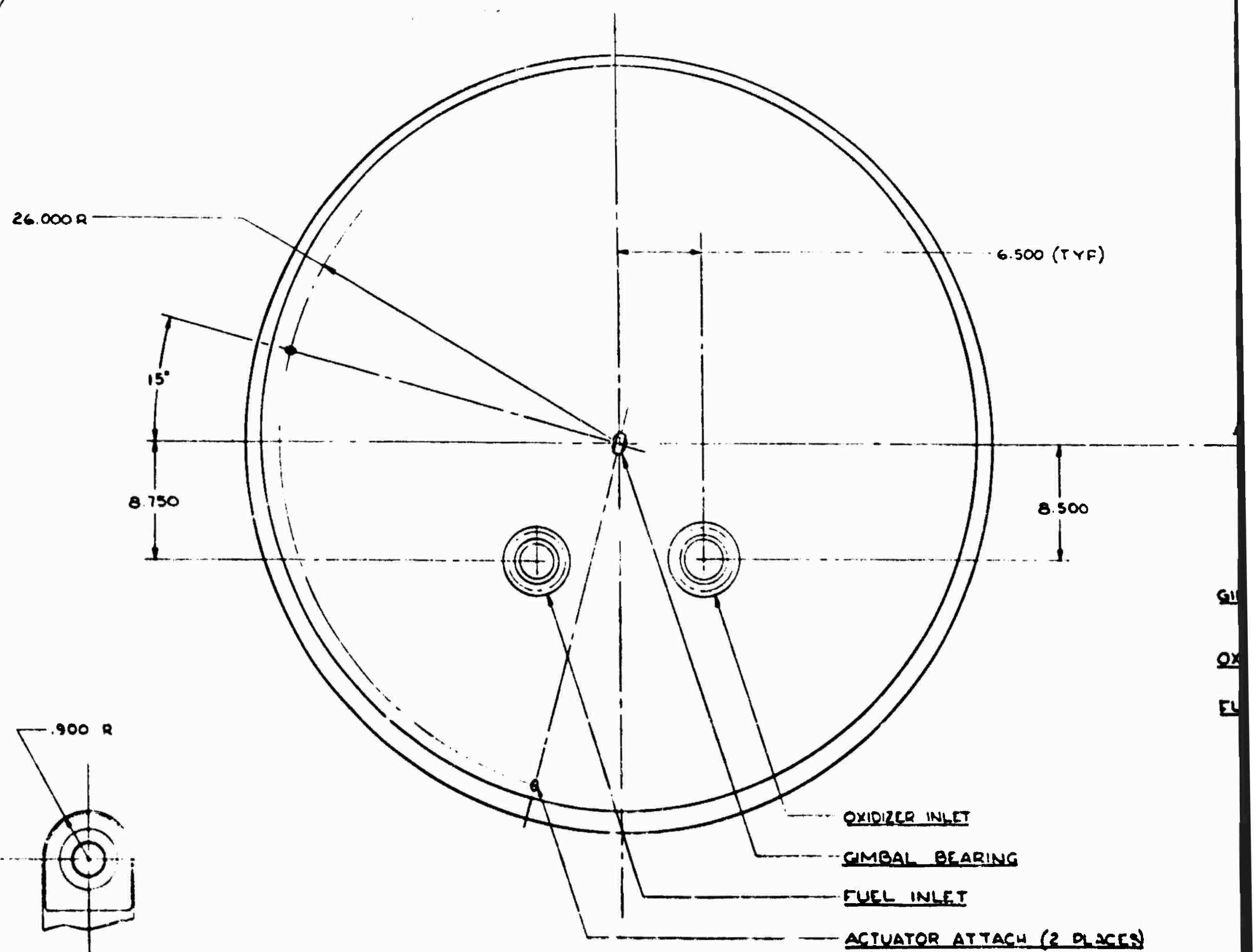
ACTUATOR ATTACHMENT
(FULL SIZE)

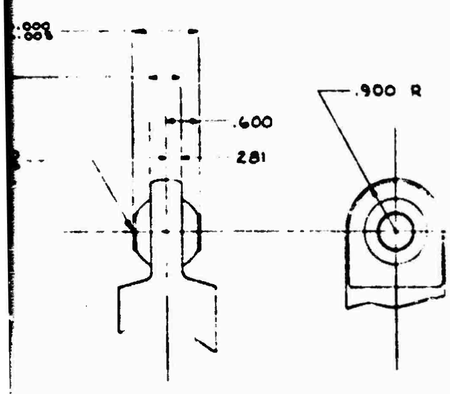


GIMBAL
(FULL



OXIDIZER INLET
(FULL SIZE)





GIMBAL BEARING
(FULL SIZE)

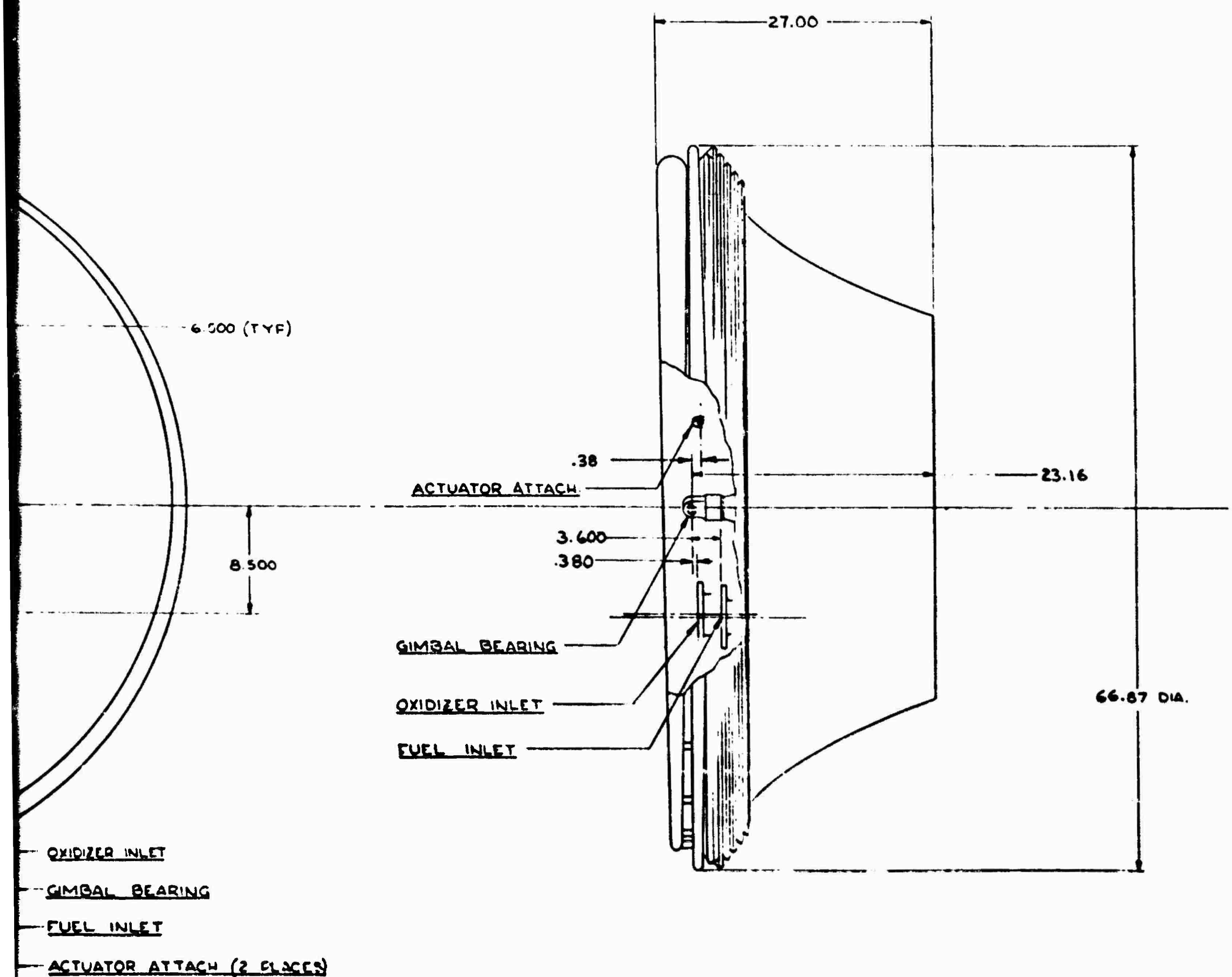


Figure 12. Engine Interface

3

TABLE 20. AEROSPIKE ENGINE SYSTEM ELECTRICAL REQUIREMENTS

Function	Volts	Current Drain, amps	Number of Components	Total Power Requirement, watts	Maximum Mission Energy Storage Requirements watt-hours	
					Mission I	Mission II
3-Way Solenoid Valve (Main Propellant Valves)	26-30 dc	3.0	2	120-180*	72	360
2-Way Solenoid Valve (Pneumatic & Purge)	26-30 dc	2.0	3	120-180**	30	130
Turbine Control Valve Actuator	110 ac***	0.3	1	25*	10	50
Turbine Bypass Control Valve Actuator	110 ac**	0.3	1	25*	10	50
Gimbal Actuator	110 ac***	0.3-5.0	2	40-590	200	1000
Engine Controller	26-30 dc	0.5	1	10-20*	8	40
Instrumentation	26-30 dc	0.5	-	10-20*	8	40
Ignition	26-30 dc	0.1-2.0	2	10-120	0.4	4
Total					338.4	1674

* Continuous power during engine firing
** Continuous power to one valve and two valves activated only during purge operation
*** Desired if available (26-30 vdc acceptable)

Mission I - 6 Starts 0.4 hour hot firing duration
Mission II - 60 Starts 2.0 hour total hot firing duration

the engine system was not considered necessary for the flight engine system, but would be included during engine development or a demonstrator engine. A schematic of the basic pneumatic system for the engine system is shown in Fig. 13 .

The estimated leakage rates for the various pneumatic system components are shown below for upstream helium conditions of 750 psig and 200 R.

Component	Estimated Leakage
Three-way solenoid valve	60 scim each
Two-way solenoid valve	2 scim each
Regulator and relief valve	50 scim total (external)
Main valve actuator	150 scim each

The leakage of the three-way solenoid valves, the two-way solenoid valves used in the purge lines, the external leakage of the regulator and relief valve, and the main valve actuators will be experienced throughout engine operation. The leakage through the on/off control portion of the two-way solenoid valve regulator is assumed lost during nonoperative periods (orbital coast).

The total leakage rate during engine operation was 470 scim. For a total engine operating duration of 0.4 hour per mission, this leakage totals 11,300 sci (0.07 lb) of helium. The orbital coast period leakage of 2 scim results in 40,300 sci (0.24 lb) of helium required for a 14-day mission duration. The pneumatic system requirement for main valve actuation is 2600 sci for each start. For a mission requiring six starts, the total helium required is 15,600 sci (0.09 lb).

The purge flow to the seal cavity of the oxidizer turbopump was continuous during engine operation. This helium purge requirement was approximately 0.0021 lb/sec, or 3 pounds for a mission requiring a total engine operating duration of 0.4 hour.

The oxidizer feed system purge operates for approximately 1 second at start and 4.5 seconds at cutoff. The feed system purge requirements were determined by the feed system volumes. The purge requirements were estimated at approximately 0.1 pound of helium per start. A chart summarizing the engine system helium requirements for two mission profiles is shown in Table 21.

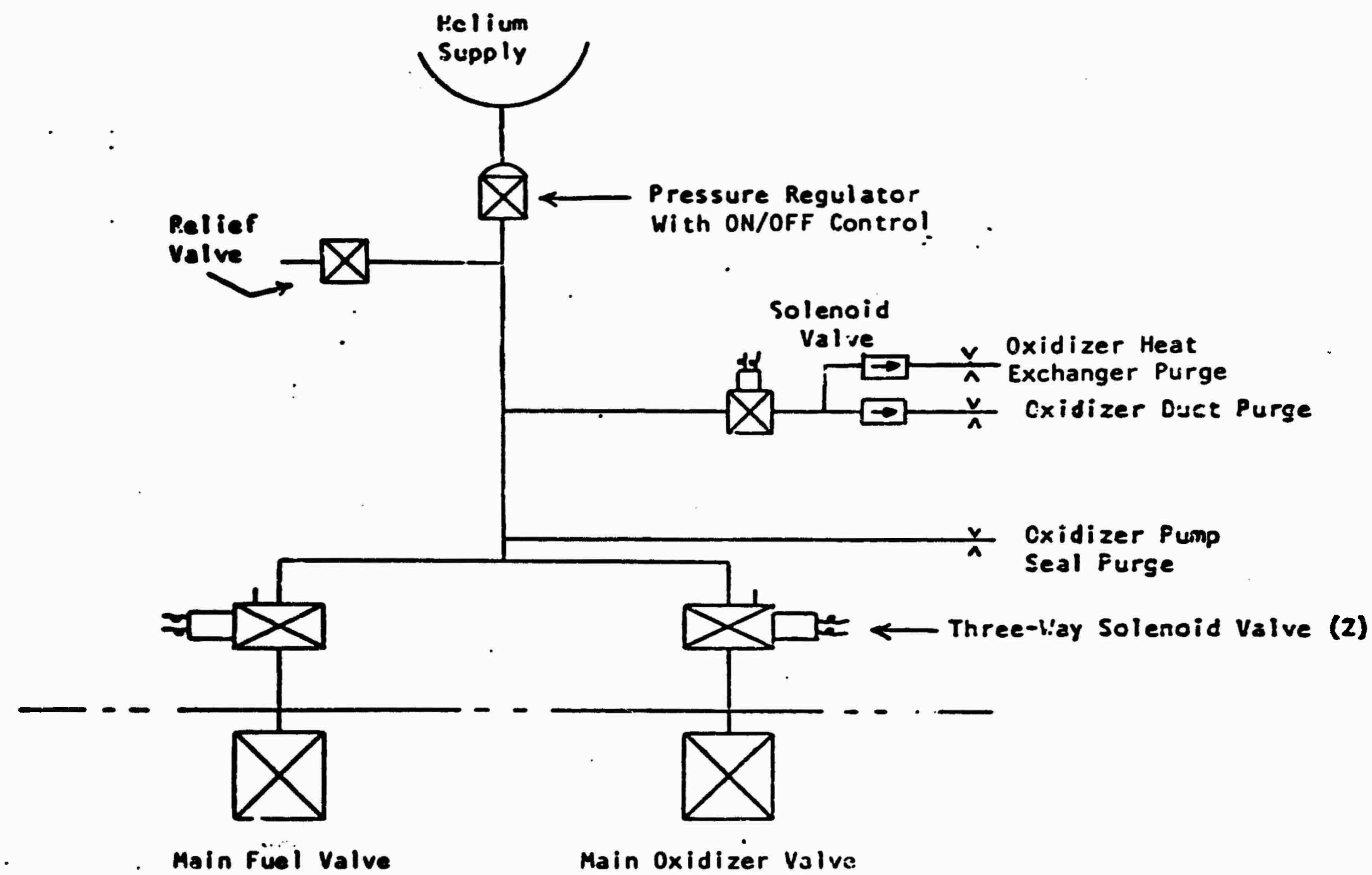


Figure 13. Pneumatic Control System

TABLE 21. AEROSPIKE ENGINE SYSTEM PNEUMATIC REQUIREMENTS

Function	Number of Components	Total Usage Requirement	Total Mission He Requirement, sci Mission I	Mission II
PNEUMATIC COMPONENT LEAKAGE				
Solenoid Isolation Valve	1	2 scim*	40,300	40,300
Regulator and Relief Valve	1	50 scim**	1,200	6,000
3-Way Solenoid Valve	2	120 scim**	2,880	14,400
Main Valve Actuator	2	300 scim**	7,200	36,000
Purge Solenoid Valve	1	2 scim**	48	240
MAIN VALVE ACTUATION	2	2600 sci/start	15,600	156,000
PUMP SEAL PURGE	2	20,000 scim**	480,000	2,400,000
OXID.SYSTEM PURGE	-	12,000 sci/engine firing	72,000	720,000
Total			619,228 (3.6 lb)	3,372,940 (20 lb)

* Continuous during orbital coast
** Continuous during engine firing

Mission I - 14 days in orbit 6 starts and 0.4 hour hot firing duration
Mission II - 14 days in orbit, 60 starts and 2.0 hours hot firing duration

Engine Instrumentation. The parameters to be monitored for safe and efficient operation and control of the engine are shown in Table 22. They consist of pressures, temperatures, flowrates, speeds, and valve displacements. The functional usage of sensor outputs is indicated in the table and comprises the following functions: engine start control, thrust control, mixture ratio control, engine limit control, engine ready, and vehicle performance evaluation.

The range, accuracy, and response of the various sensors is shown in Table 22. Dual redundant sensors and integrated pressure – temperature transducers are used to achieve savings in system weight and to improve reliability. Three outputs from dual redundant sensors are provided to the controller for thrust and mixture ratio control.

The controller tests all sensor outputs for consistency with reference values and for consistency with one another, then averages all three sensor outputs to achieve desired precision. In the event of failure of one or two sensor outputs, outputs from alternate sensors are used (Table 23).

Pressures. The pressure sensor design used the basic sensing element developed for the Saturn program. Sensors consist of a diaphragm connected by a link pin to a deflecting beam incorporating strain gages, electrically connected to form Wheatstone bridge circuits. Each beam contains two fully active bridge circuits for redundancy and a set of resistors in each bridge for electrical checkout.

Sensors are mounted directly at the point of measurement, thus eliminating pressure-sensing lines. This has been made possible by locating the signal conditioning electronics at the controller. Pressure transducer accuracy is ±2 percent of full-scale reading. Calibration of the transducer prior to installation in the engine is made to attain precision limits of ±0.5 percent.

Temperatures. System cryogenic temperatures are sensed by resistance-type temperature transducers mounted at the point of measurement. Each sensor contains

TABLE 22. FLIGHT INSTRUMENTATION

	Function *	Range, psia	Transducer Accuracy %	Response HZ
Pressures:				
Thrust Chamber	1 2 3 4	0-1000	± 2	0-100
T/C Coolant Jacket Exit	4	0-2000	± 2	0-100
Fuel Turbopump Discharge	1 4	0-3000	± 2	0-100
Oxidizer Turbopump Discharge	1 4	0-1500	± 2	0-100
Fuel Turbine Inlet	4	0-2000	± 2	0-100
Oxidizer Turbine Inlet	4	0-2000	± 2	0-100
Fuel Injection	4	0-1500	± 2	0-100
Fuel Flowmeter Inlet	1 2 3 4	0-3000	± 2	0-100
Oxidizer Flowmeter Inlet	1 2 3 4	0-1500	± 2	0-100
Fuel Turbopump Inlet	4	0-300	± 2	0-100
Oxidizer Turbopump Inlet	4	0-300	± 2	0-100
Helium Tank	4 5	0-3000	± 2	0-100
Temperatures:		°F	%	sec
Main Fuel Injection	4	-423 to + 700	± 2	0.2
Thrust Chamber Jacket (Flowmeter) Inlet	1 2 3 4	-423 to + 700	± 2	0.2
Thrust Chamber Jacket Outlet	4	-423 to + 700	± 2	0.2
Thrust Chamber Skin	4	-423 to + 700	± 2	0.2
Fuel Turbopump Discharge	4	-423 to + 700	± 2	0.2
Oxidizer Turbopump Discharge	1 4	-423 to + 700	± 2	0.2
Fuel Turbine Inlet	4	-423 to + 700	± 2	0.2
Oxidizer Turbine Inlet	4	-423 to + 700	± 2	0.2
Oxidizer Flowmeter Inlet	1 2 3 4	-423 to + 700	± 2	0.2
Fuel Pump Inlet	4	-423 to + 700	± 2	0.2
Oxidizer Pump Inlet	4	-423 to + 700	± 2	0.2
Helium Tank	4 5	-423 to + 700	± 2	0.2

TABLE 22. (Concluded)

	Function*	Range lb/sec	Transducer Accuracy %	Response rad./sec
Flowrates:				
Engine Main Fuel	1 2 3 4	0-10	± 0.15	300
Engine Main Oxidizer	1 2 3 4	0-50	± 0.15	300
Valve Position:		degrees	%	-
Main Fuel Valve	1 4 5	0-90	± 5	-
Main Oxidizer Valve	1 4 5	0-90	± 5	-
Turbine Bypass Valve	1 4 5	0-90	± 5	-
Shaft Speeds:			rpm	
Fuel Turbopump	4 6		± 1	-
Oxidizer Turbopump	4 6		± 1	-

*Function:
1 Engine Start Control
2 Engine Thrust Control
3 Mixture Ratio Control
4 Vehicle Performance Evaluation
5 Engine Ready
6 Engine Limit Control

TABLE 23. ALTERNATE CONTROL SENSORS

PERFORMANCE CONTROL PARAMETER	ALTERNATE SENSOR VALUE COMPUTED FROM
Main combustion chamber pressure	Mixture ratio and total fuel and oxidizer flowrates.
Fuel flowmeter inlet pressure	Fuel turbopump speed and fuel flowrate.
Fuel flowmeter inlet temperature	Constant.
Fuel turbopump flowrate	Main combustion chamber pressure and oxidizer flowrate.
Oxidizer turbopump flowrate	Main combustion chamber pressure and fuel flowrate.
Oxidizer flowmeter inlet pressure	Oxidizer turbopump discharge pressure and speed.
Oxidizer flowmeter inlet temperature	Constant.

two platinum wire-wound elements which follow resistance-versus-temperature curves as defined by the National Bureau of Standards. Dual elements provide redundancy and minimize the number of sensor ports on the engine.

The resistors for completing the bridge and the electronics for signal conditioning and checkout are located in the controller. The electrical harness wiring resistance from the controller to the element is nullified in the resistance bridges by using a system of three harness electrical wires to each element.

Flowrates. Flowrates are measured with turbine-type flowmeters designed for high-velocity flow, low drag, and minimum pressure drop, as used on the Saturn J-2 engines. These flowmeters have demonstrated high accuracy and reliability limits. The design selected incorporates a spring-actuated, pneumatically deactivated brake shoe which acts on a shroud surface placed over the turbine blades. The brake prevents uncontrolled flowmeter spinning during checkout and purging, which might result in damage to the uncooled bearings. The flowmeter speed is monitored by magnetic pickups. Pressure and temperature measurements for use in calculating flow weight ratio are obtained with the use of an integrated pressure-temperature transducer unit. This unit contains two redundant sets of temperature and pressure sensors.

Rotating Speeds. The pump shaft speed and flowmeter speed sensors are of the magnetic pickup type. Each pickup contains two coils and produces dual signals. The interface with the controller is through the electrical harness. Pump and flowmeter speeds are monitored by pulse rate counters in the controller.

Valve Positions. The valve position indicators are an integral part of all the valves. These indicators are used for inner loop control feedback in the servovalves, for engine ready signals, and for engine start. Two outputs from dual redundant sensors are provided on all servovalves to improve reliability. Main propellant valves are provided with single sensors.

SINGLE-PANEL ENGINE SYSTEM DESCRIPTION

The baseline single-panel, O_2/H_2 AMPS engine system (shown in Fig. 14) is based on an aerospike thrust chamber design with a maximum thrust level of 25,000 pounds and a throttling capability to 5,000 pounds. The maximum operating chamber pressure is 750 psia and the aerospike nozzle has an expansion area ratio of 110:1. The nominal operating engine mixture ratio is 5.5:1 with an off-design operational capability of 0.5 mixture ratio units for propellant utilization purposes. The thrust chamber and nozzle are regeneratively cooled with the entire fuel flowrate; thus, the combustion chamber injector is fed with heated, gaseous hydrogen and liquid oxygen. The injector incorporates a concentric orifice element configuration.

The engine is pump fed by centrifugal-type pumps, which are directly driven by axial flow impulse turbines . The expander topping cycle was selected as a superior turbopump drive cycle for this engine based primarily on performance and engine system simplicity. The heated hydrogen, after passing through the thrust chamber regenerative cooling circuit, is fed to the two turbines in a parallel flow arrangement. The turbine exhaust flow, except for a small flow which is directed to the base of the aerospike nozzle (approximately 0.2 percent of the total thrust chamber flow), is then directed to the injector fuel inlet manifold.

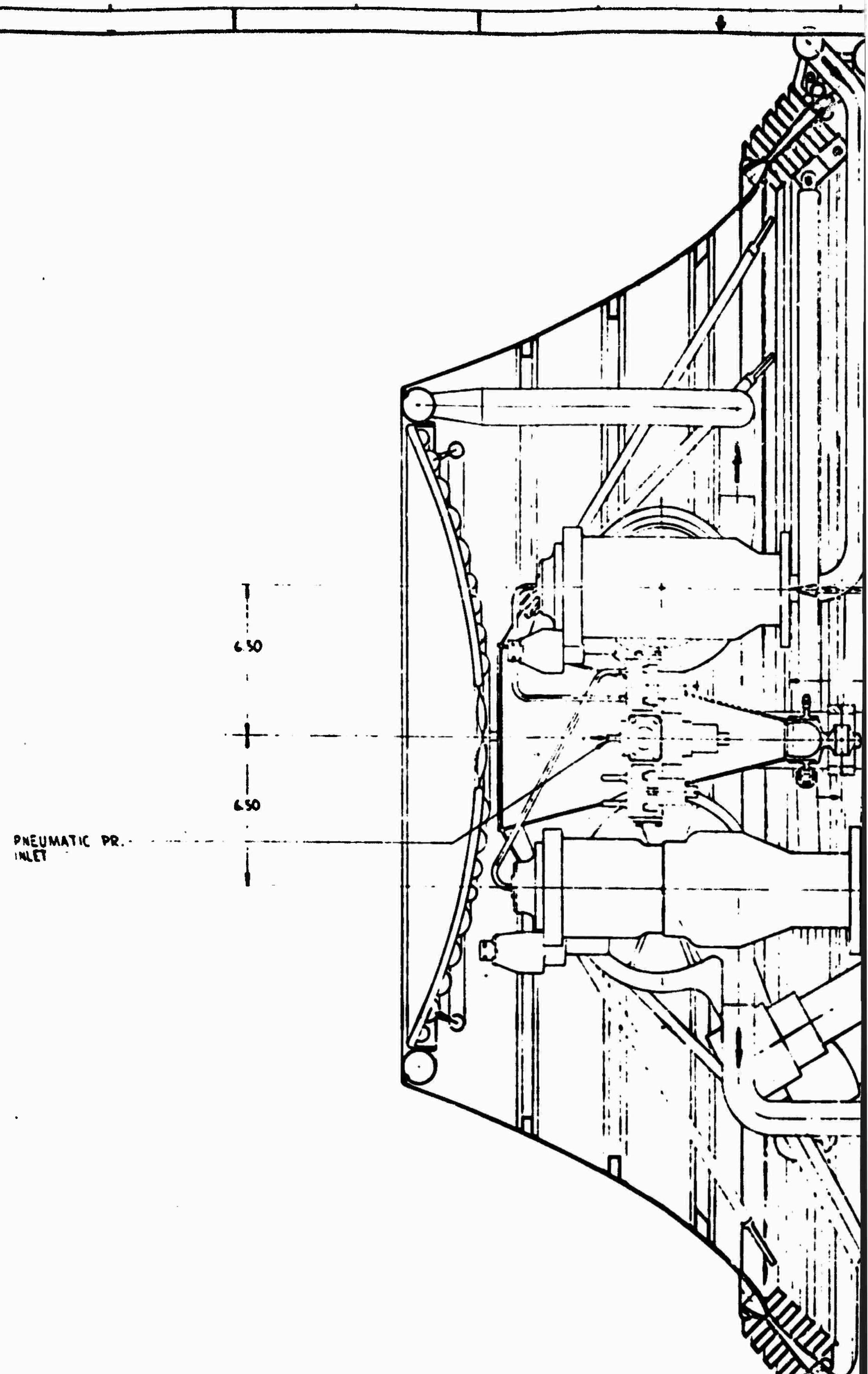
The engine control system consists of main propellant valves (two liquid valves) located upstream of the turbopumps, an oxidizer turbine inlet control valve, and a turbine bypass valve (two hot hydrogen valves). The propellant main valves are pneumatically actuated and the variable-area turbine control valves are electrically actuated. The turbine bypass valve provides thrust control by varying the amount of heated hydrogen that passes through the turbines. An adequate power and control margin is provided at the nominal full-thrust operating point by designing for a nominal bypass flow equal to 20 percent of the total fuel flow. Engine mixture ratio control capability is provided by the oxidizer turbine inlet valve. Engine operation is controlled by a system which receives guidance system commands and engine parameter feedback, and then computes the engine control signals.

The engine was designed for multiple starts at altitude. Engine start is accomplished through a propellant tank-head start sequence. The hydrogen was pressure fed from the main tanks through the thrust chamber regenerative cooling jacket, where the hydrogen temperature is increased due to the residual heat capacity of the chamber. The warm hydrogen then passes through the turbines, providing the initial power to the fuel and oxidizer pumps according to the prescribed sequence. After ignition, the engine bootstraps to full power.

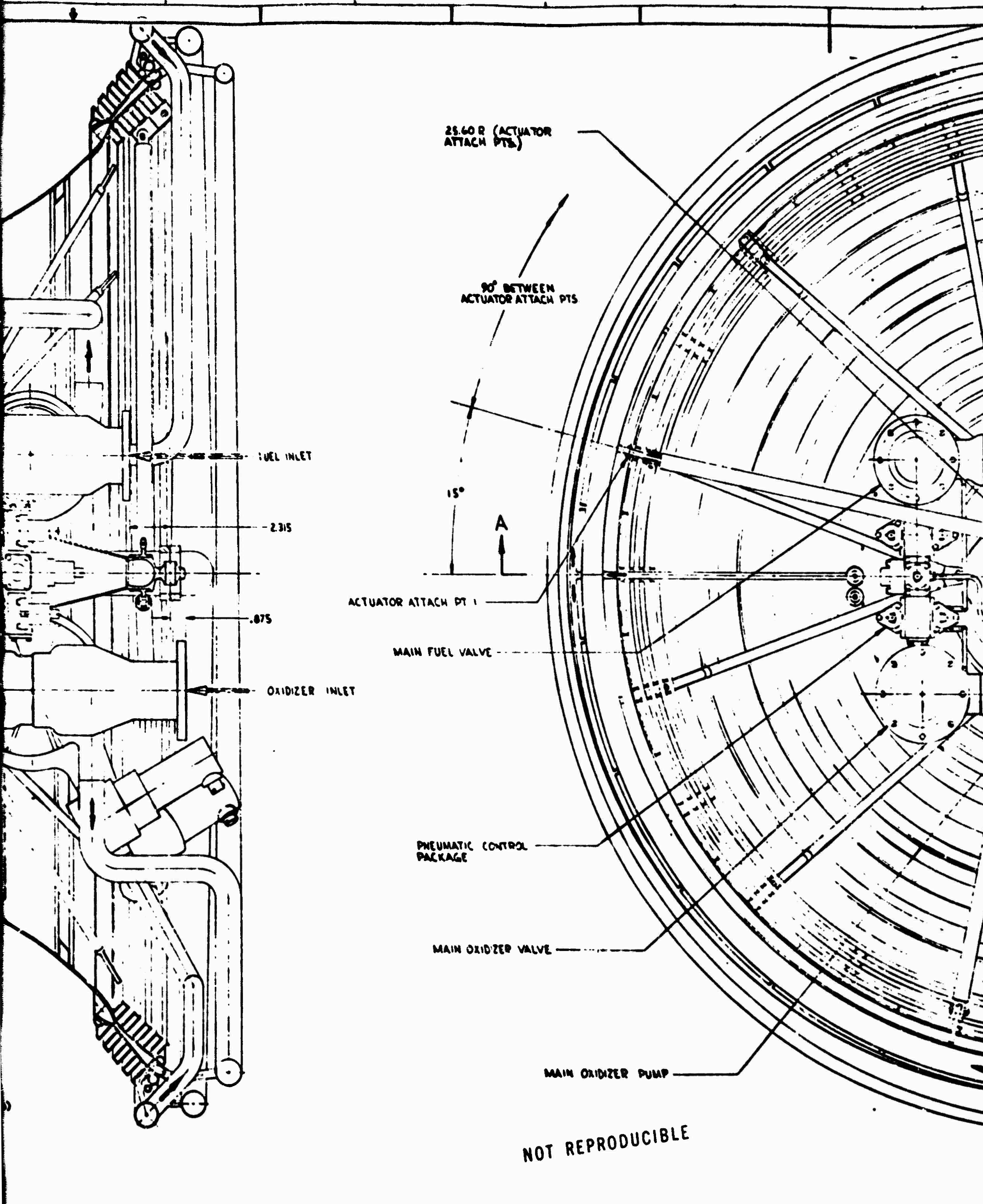
A combustion wave ignition system is used where a spark-induced combustion wave passes through an unburned, gaseous oxygen/hydrogen mixture to ignite a pilot element within each combustion chamber segment. A central mixing chamber/gas generator is used to mix and distribute the propellants through the delivery tubes to each segment. The combustion wave is initiated by an electrical arc discharge spark in the premix chamber at the same time that the oxidizer flow to the premix chamber is cut off. The resulting combustion wave then propagates in the unburned mixed propellants toward the segments. Fuel is fed to the premix chamber from the cooling jacket exit and oxidizer is fed from the pump discharge under tank pressure for the initial portion of the start transient.

The engine utilizes helium gas for main valve actuation, oxidizer turbopump seal cavity purge, and propellant system surges.

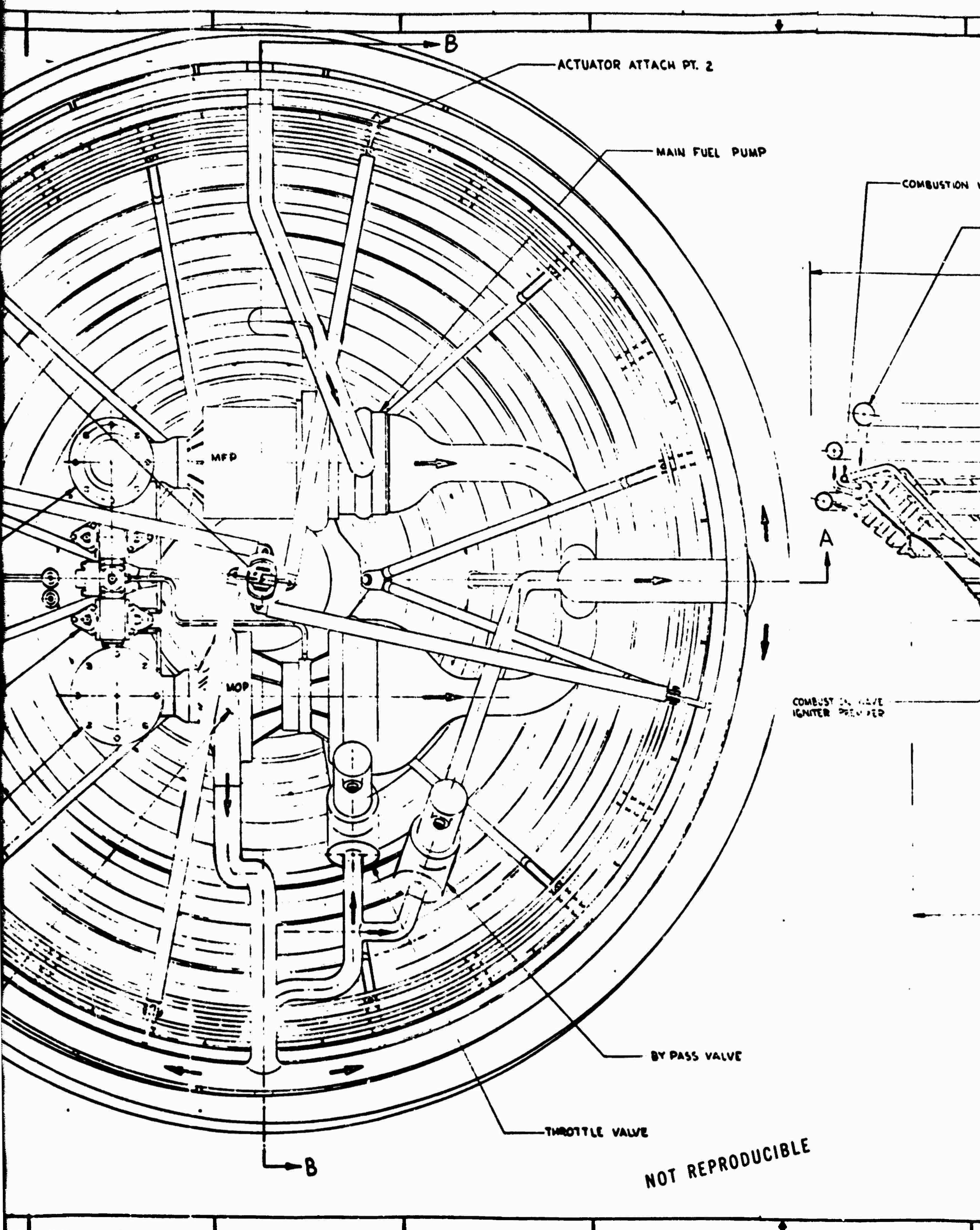
The basic engine system design parameters are shown in Table 24. Delivered engine performance over the thrust and mixture ratio range is shown in Fig. 15. An engine system propellant flow schematic is shown in Fig. 16.

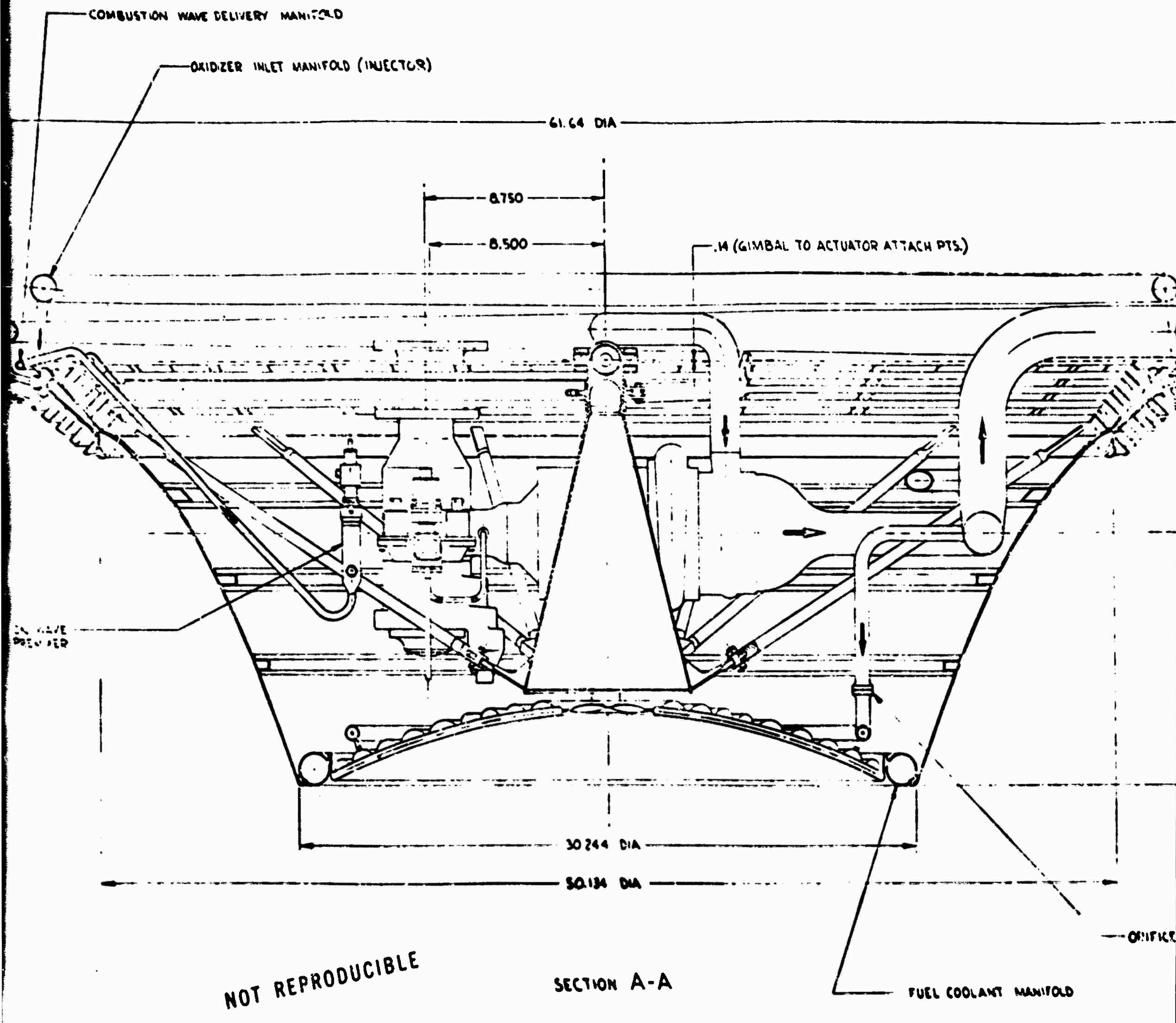


SECTION B-B
(PUMPS/VALVES NOT SECTIONED)



NOT REPRODUCIBLE


B
ACTUATOR ATTACH PT. 2
MAIN FUEL PUMP
COMBUSTION W
MFP
MOP
A
BY PASS VALVE
THROTTLE VALVE
B



NOT REPRODUCIBLE

SECTION A-A

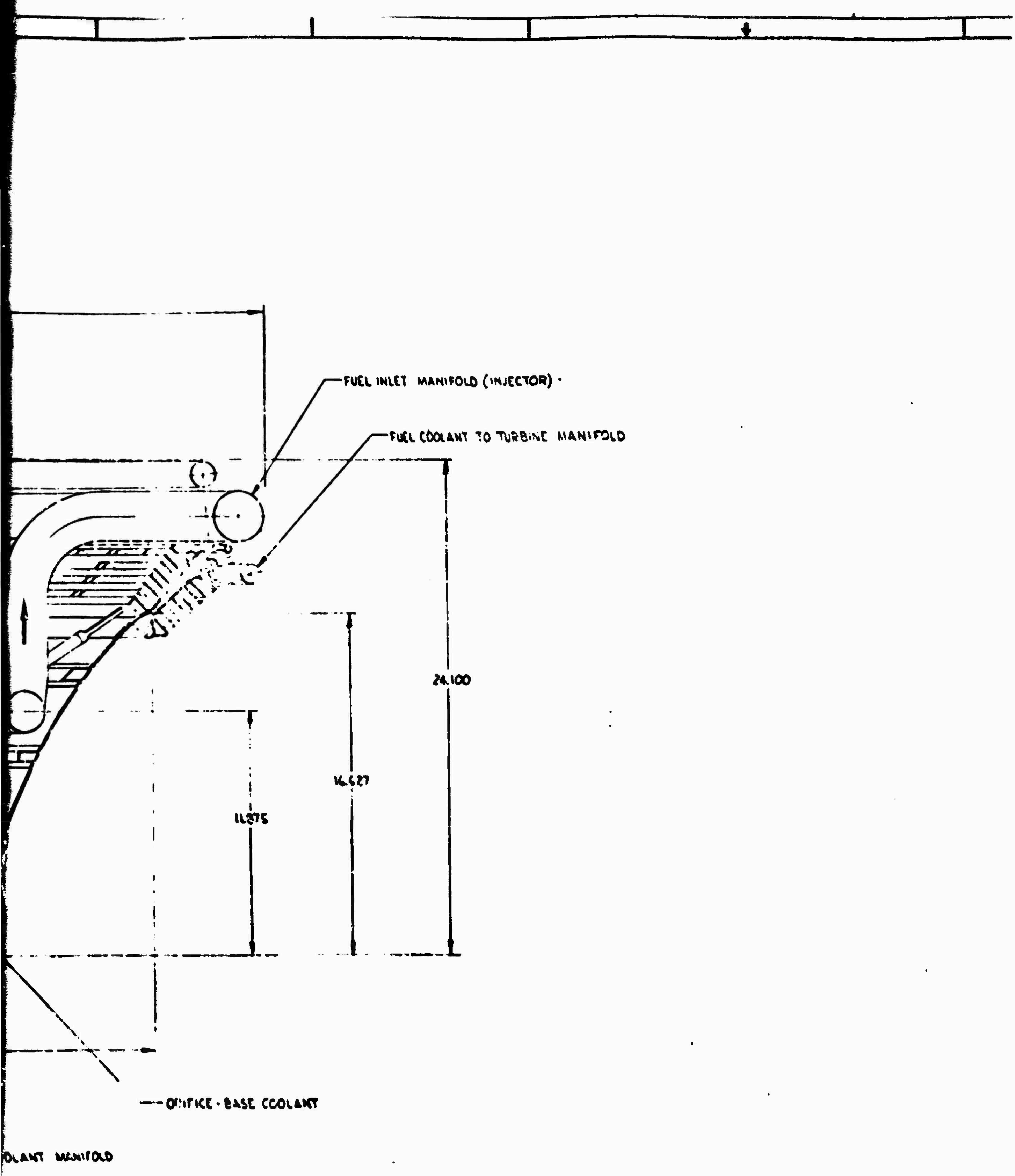


Figure 14. 25,000-Pound-Thrust, Single-Panel Aerospike Engine System

5

TABLE 24. AEROSPIKE ENGINE SYSTEM DESIGN AND OPERATING CONDITIONS (SINGLE-PANEL COOLING)

Thrust at Vacuum, pounds	25,000
Nozzle Stagnation Pressure, psia	750
Expansion Area Ratio	110:1
Total Oxidizer Flowrate, lb/sec	46.187
Total Fuel Flowrate, lb/sec	8.398
Engine Mixture Ratio	5.5:1
Thrust Chamber Mixture Ratio	5.57:1
Specific Impulse, seconds	458.0
Nozzle Base Flow, lb/sec	0.109
Hydrogen Injection Temperature, °R	808
Turbine Inlet Temperature, °R	839
Percent Turbine Bypass	20
Turbine Inlet Pressure, psia (Fuel/Oxidizer)	1156/ 116
Turbine Pressure Ratio (Fuel/Oxidizer)	1.30/1.24
Fuel Pump Discharge Pressure, psia	1591
Oxidizer Pump Discharge Pressure, psia	1062
Engine Length, inches	24
Engine Diameter, inches	62

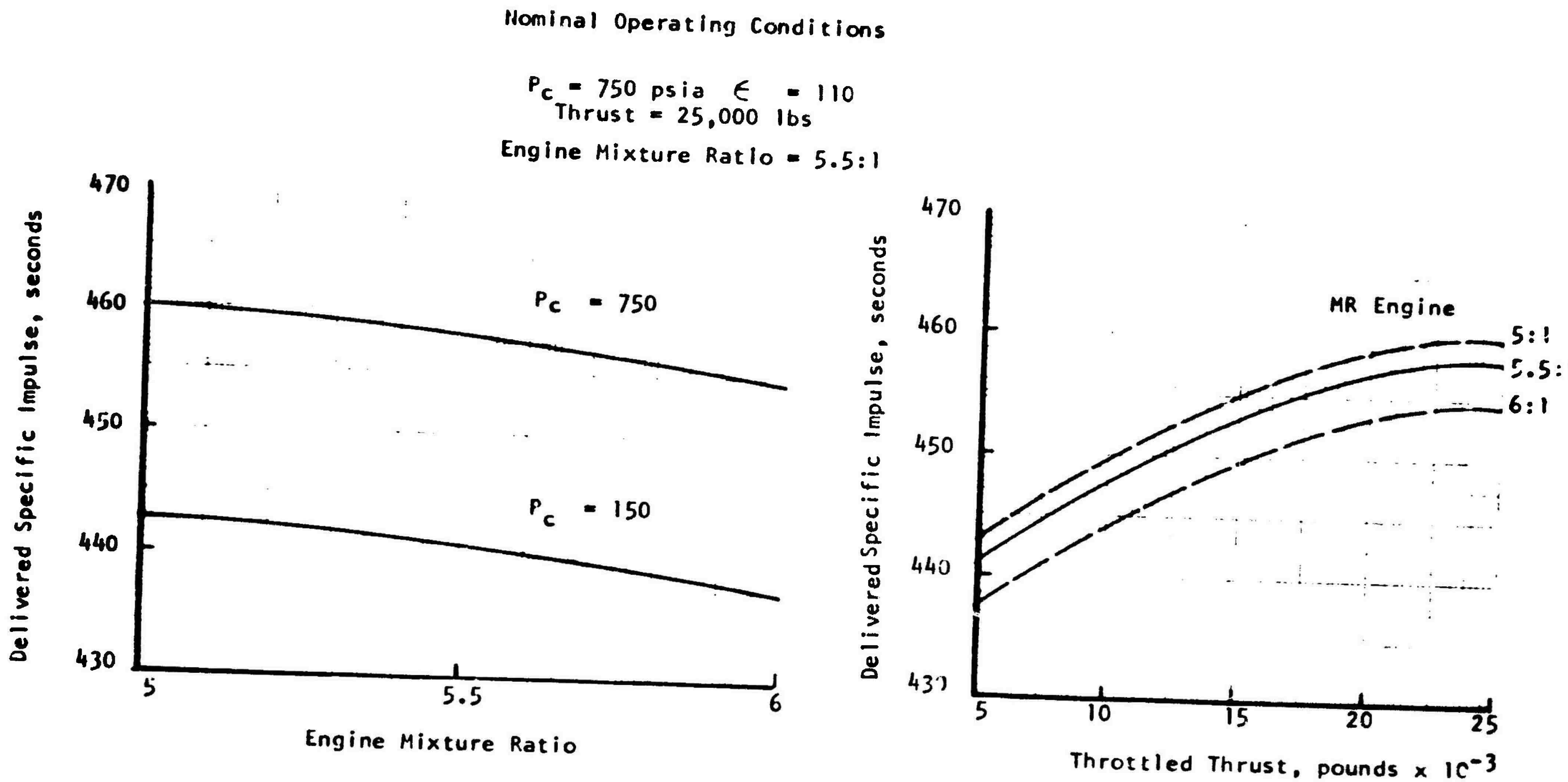
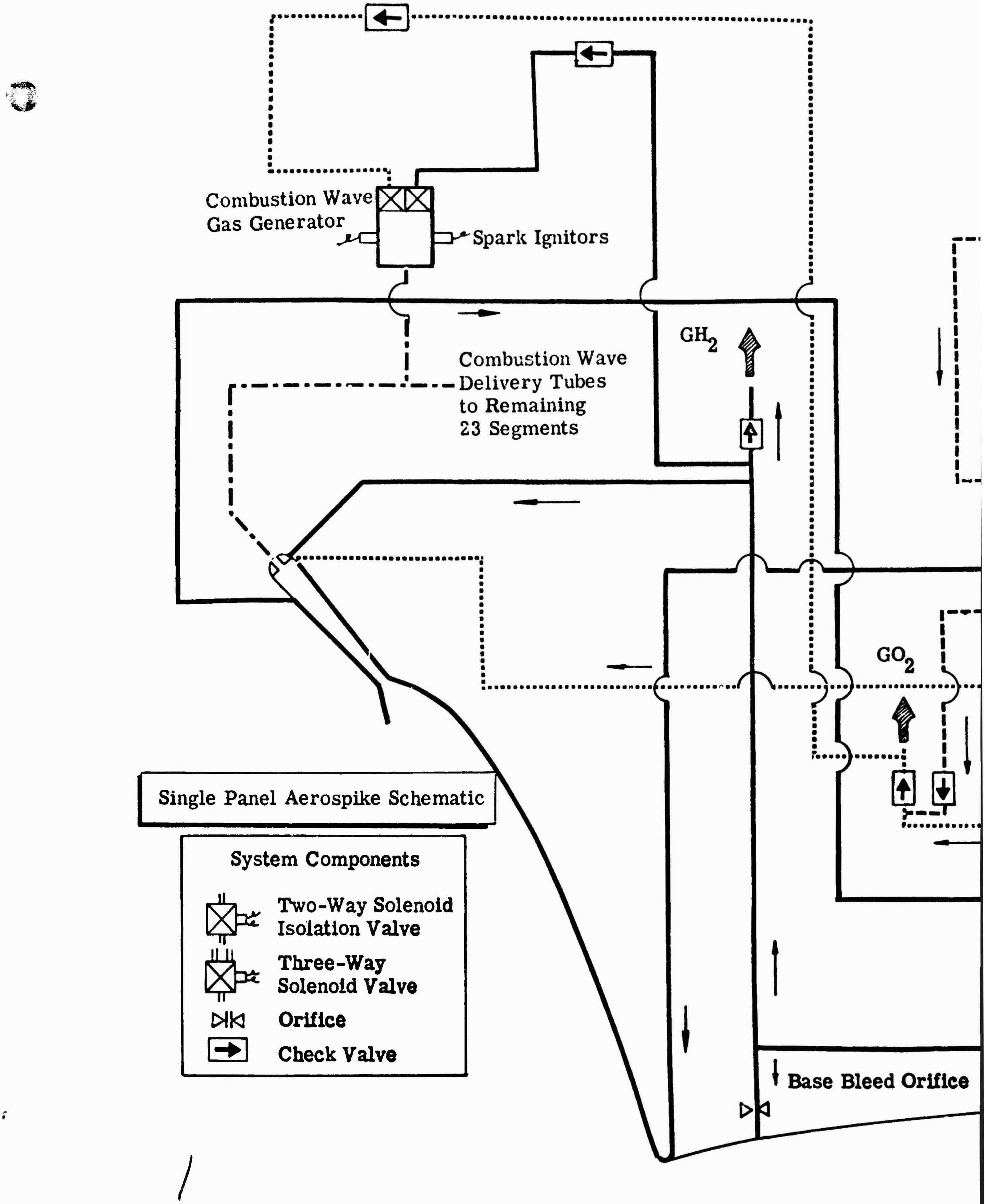


Figure 15. Baseline Single-Panel Aerospike Engine Off-Design Performance



Single Panel Aerospike Schematic

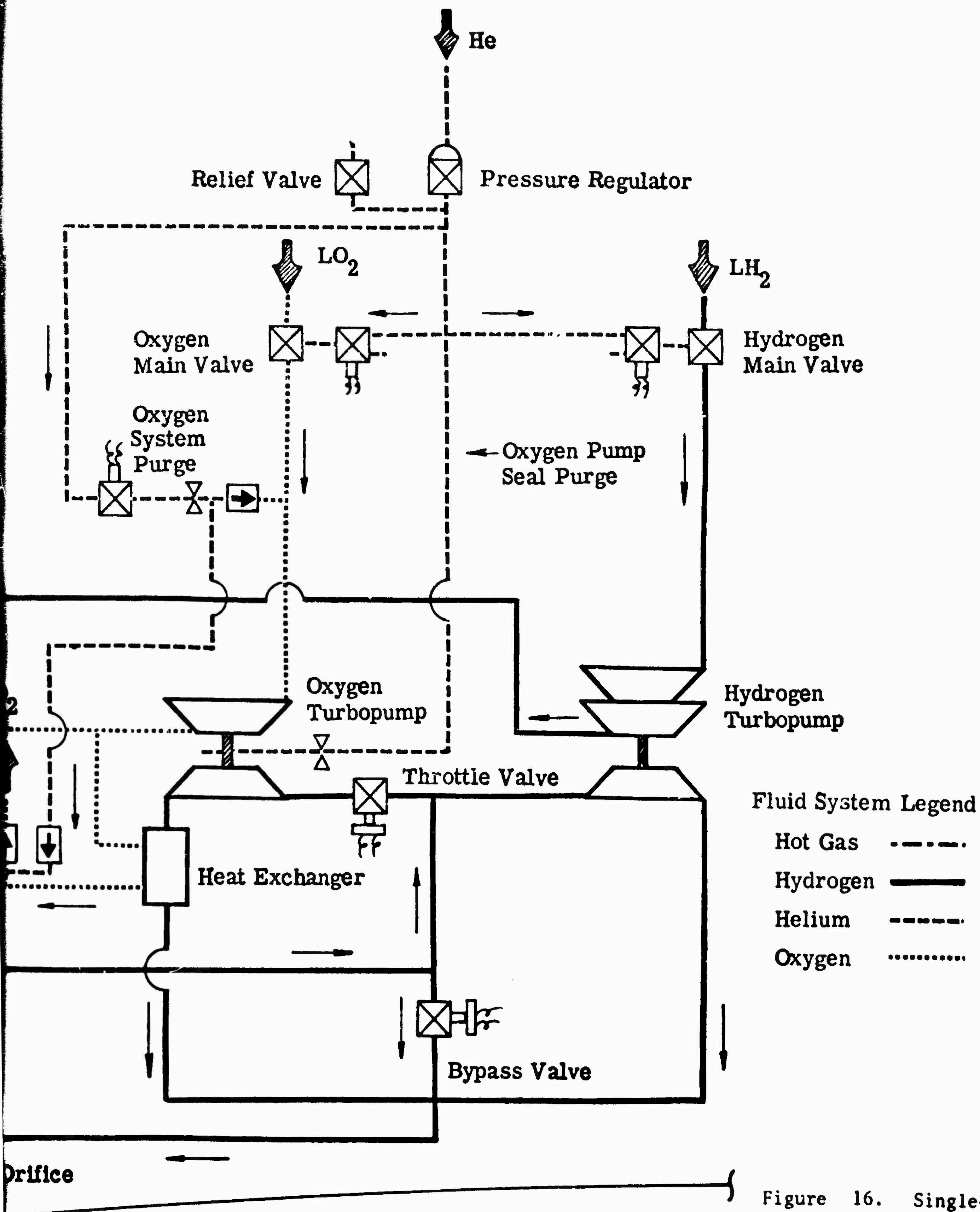


Figure 16. Single-Panel Aerospike Engine Schematic

2

Engine System Weight and Mass Properties

The component weights for the baseline single-panel, 25,000-pound-thrust aerospike engine system are shown in Table 25. The combustion chamber design incorporates the single-panel cooling circuit and brazed-on copper sheet closeouts to the segment castings. A contingency weight of 29 pounds was included to account for uncertainties in the weight estimates of those components that have not as yet received detailed design attention and possible future production weight growths.

Engine Interface Requirements

Structural, fluidic, and electrical interface connections between the engine system and propellant feed system are required. The locations and dimensions of the structural mountings and main propellant inlets to the engine system are shown in Fig. 17. Structural connection for transmission of thrust to the propellant feed system is made at the forward face of the gimbal mount. Gimbal actuator attachment is made at two locations 90 degrees apart where the thrust structure attaches to the thrust chamber. Flexibility in the fuel and oxidizer propellant inlet ducts for engine gimbaling, thermal growth, and manufacturing misalignment was provided on the propellant feed system side of the interface.

Engine System Electrical Requirements. Electrical requirements are not significantly different for the single- and double-panel engine systems. The requirements are discussed on page 58 in the section describing the double-panel engine design.

Pneumatic Requirements. Pneumatic requirements are not significantly different for the single- and double-panel engine systems. The requirements are discussed on page 58 in the section describing the double-panel engine design.

Engine Instrumentation. Instrumentation is not significantly different for the single- and double-panel engine systems. A discussion is presented on page 64 in the section describing the double-panel engine design.

TABLE 25. 25,000-POUND-THRUST BASELINE SINGLE-PANEL AEROSPIKE ENGINE SYSTEM WEIGHTS

Thrust, lbs	**25,000**
Chamber Pressure, psia	**750**
Expansion Area Ratio	**110:1**
Nozzle % Length	**20**
Subsystem Weights	
Combustion Chamber and Shroud	118
Nozzle	30
Base Closure	6
Thrust Mount and Gimbal Assembly	16
Turbopumps and Mounts	60
Propellant Ducting and Inlet Valves	58
Hot-Gas Valves	16
Controls and Miscellaneous	20
Ignition System	7
Contingency	29
Total Engine System Weight	360

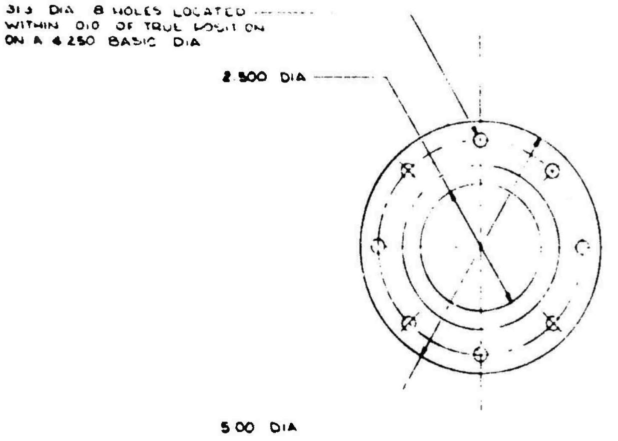
ENGINE OPERATIONAL ANALYSIS

ENGINE SYSTEM PERFORMANCE

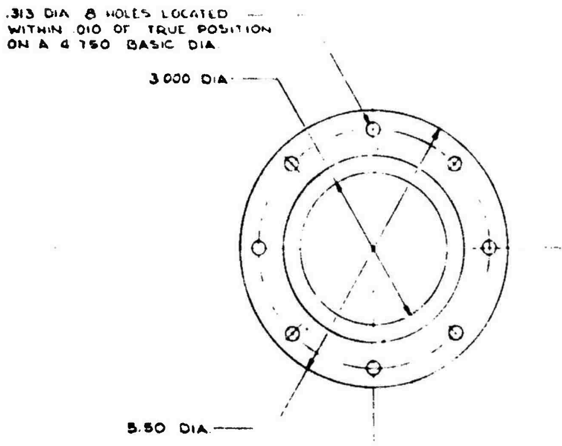
Delivered engine system performance was determined for the operational range of thrust and mixture ratio (Fig. 18). The steady-state performance computations were performed using a digital computer nonlinear mathematical engine model prepared specifically for the double-panel aerospike engine. This model accepts input performance descriptions of components such as turbines, pumps, heat exchangers, thrust chambers, lines, valves, etc. Many component descriptions are in parametric form and are the result of executions of other nonlinear models established to analyze that particular item. Solution of an engine balance point, given a set of operating conditions, is performed by iteration of selected parameters until suitable accuracy of the assumed and computer values has been achieved. Results are presented for the double-panel engine. Single-panel engine trends would be similar.

The engine specific impulse was calculated using as much of the methodology outlined in Addendum 40.1 to CPIA 178 as was applicable. Performance parameters are shown in Table 26 for the nominal operating conditions.

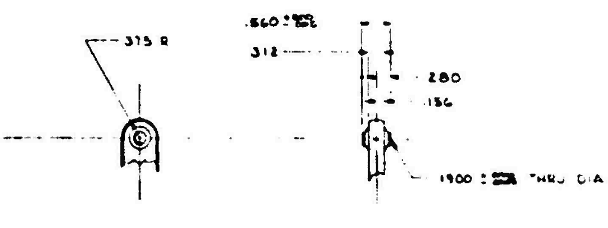
Steady-state performance balances for the double-panel aerospike engine over the specified range of operation are shown in Table 27. Balances are presented for engine mixture ratios of 5.0, 5.5 (nominal), and 6.0 at the operating thrust levels of 25,000 pounds (nominal) and 5,000 pounds. Similar data are presented in schematic form (Fig. 19 through 21) for thrust levels of 25,000, 15,000, and 5,000 pounds, at engine mixture ratio of 5.5. Parameters indicated are pressures, temperatures, and flows. Propellant leakages during engine operation are included in the balances. Fixed tank pressurization flows of 0.26 lb/sec gaseous oxygen and 0.05 lb/sec gaseous hydrogen also are included. Tank pressurization flows are included in engine mixture ratio determination, but excluded from specific impulse determination.



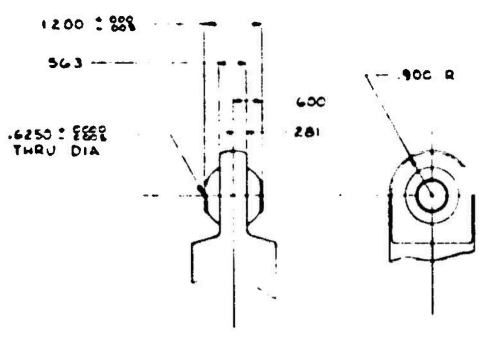
FUEL INLET
(FULL SIZE)



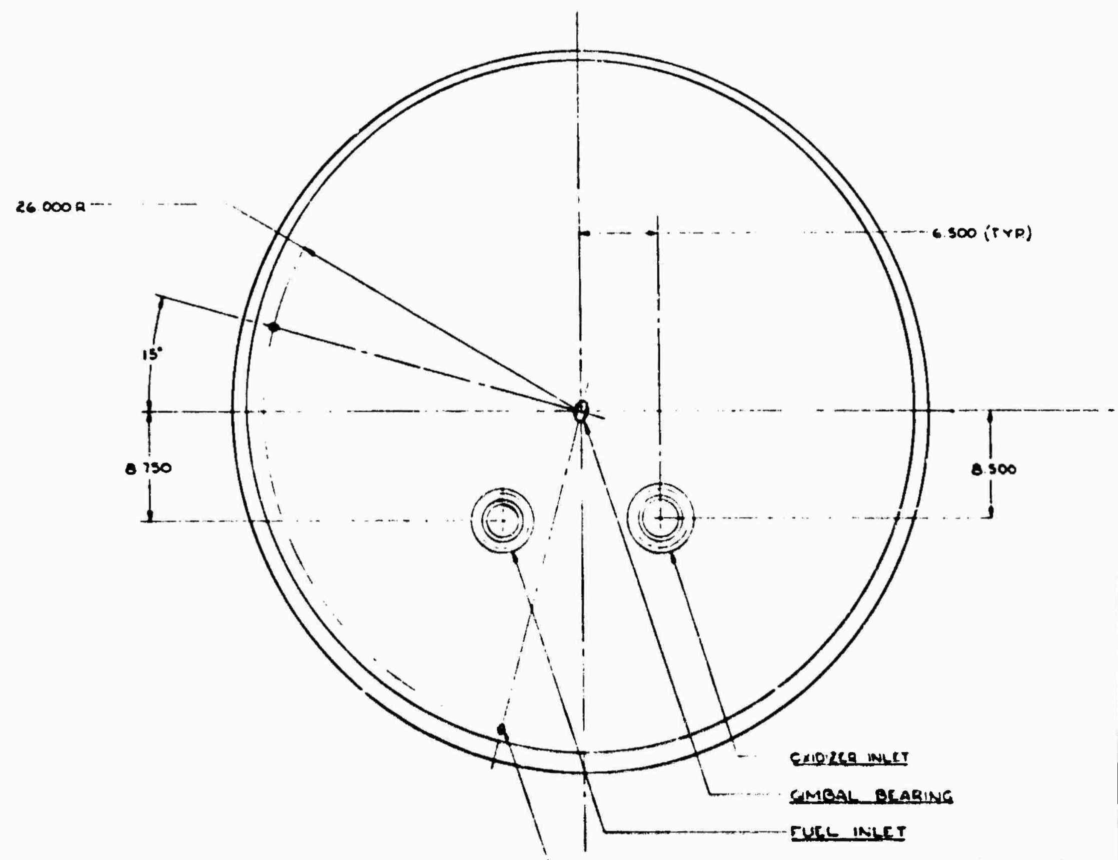
OXIDIZER INLET
(FULL SIZE)



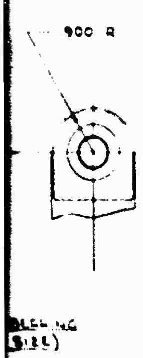
ACTUATOR ATTACHMENT
(FULL SIZE)



GIMBAL BEARING
(FULL SIZE)



.900 R
26.000 R
6.500 (TYP)
15°
8.750
8.500
OXIDIZER INLET
GIMBAL BEARING
FUEL INLET
ACTUATOR ATTACH (2 PLACES)



2

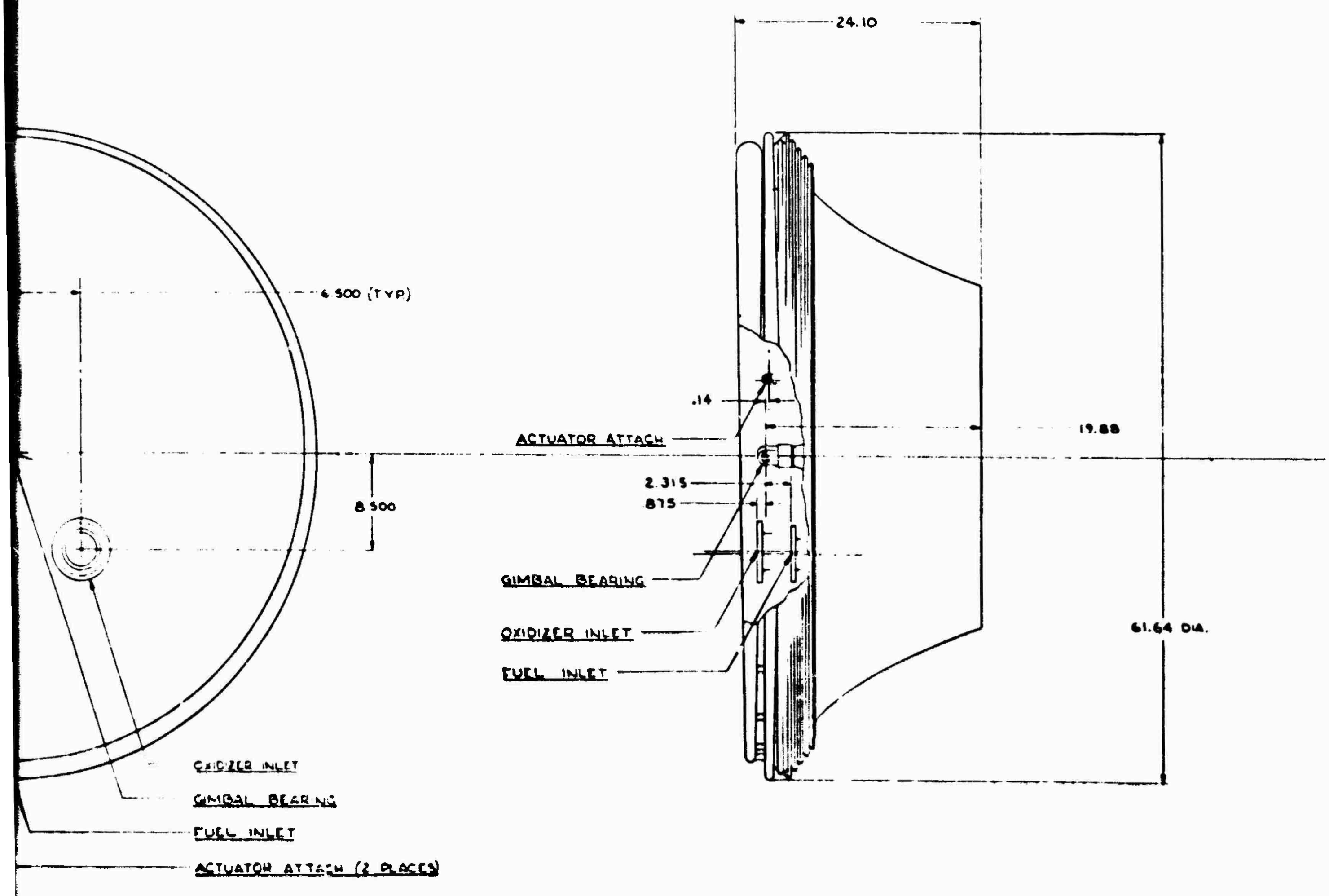


Figure 17. 25,000-Pound-Thrust Single-Panel Interface Locations

3

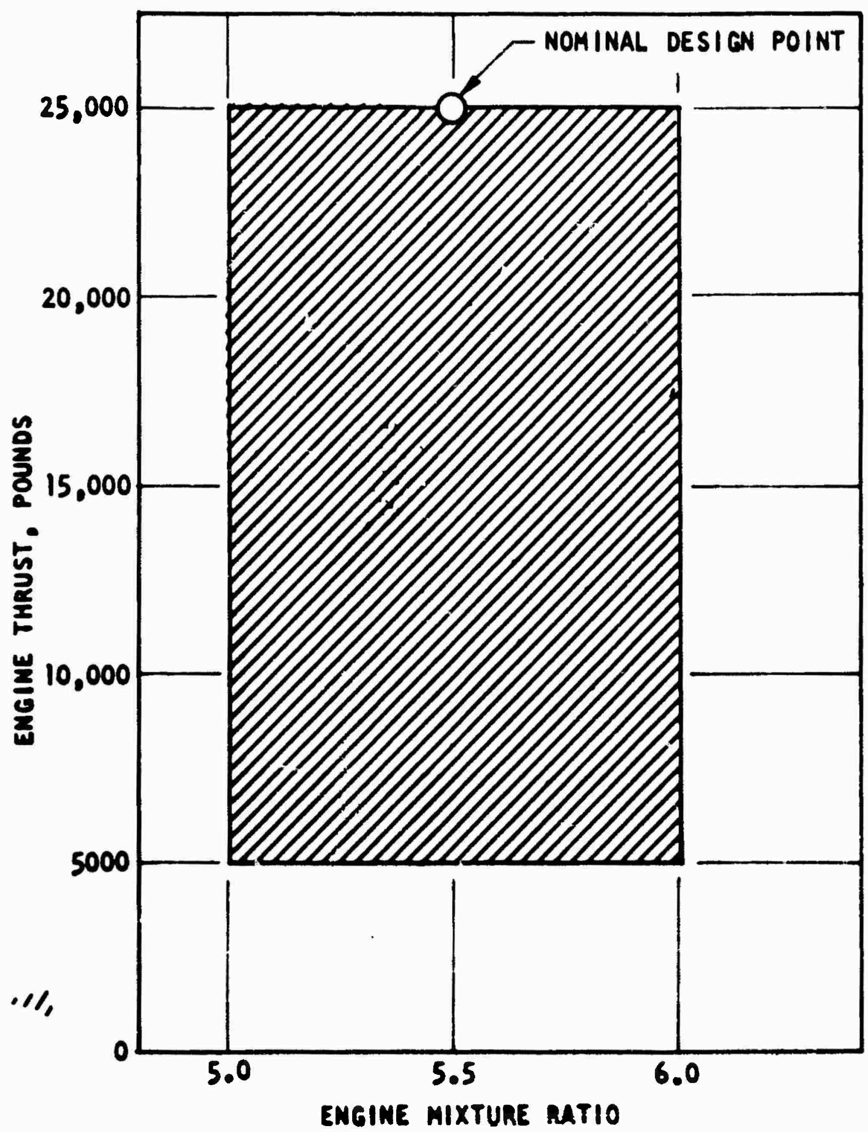


Figure 18. Aerospike Engine Thrust-Mixture Ratio Operating Envelope

TABLE 26. NOMINAL PERFORMANCE, DOUBLE-PANEL AEROSPIKE ENGINE

Expander Cycle	
Nozzle Type	Aerospike
Engine Thrust, pounds	25000
Engine Mixture Ratio	5.5:1
Area Ratio	200:1
Stagnation Pressure, psia	1000
Injector Mixture Ratio	5.572
Injector Flowrate, lbm/sec	52.99
Hydrogen Injection Enthalpy, Kcal/mole	2.18
Oxygen Injection Enthalpy, Kcal/mole	-1.005
ODIE Specific Impulse, lbf-sec/lbm	498.5
ODK Specific Impulse, lbf-sec/lbm	497.3
Divergence Efficiency	0.9671
Boundary Layer Loss, lbf-sec/lbm*	-17.93
Energy Release Efficiency	0.995
Secondary Mixture Ratio	0.0
Secondary Specific Impulse**, lbf-sec/lbm	6056.2
Secondary Flow Ratio, $\dot{M}_{secondary}/\dot{M}_{primary}$	0.0019
Engine Delivered Specific Impulse, lbf-sec/lbm	470.4

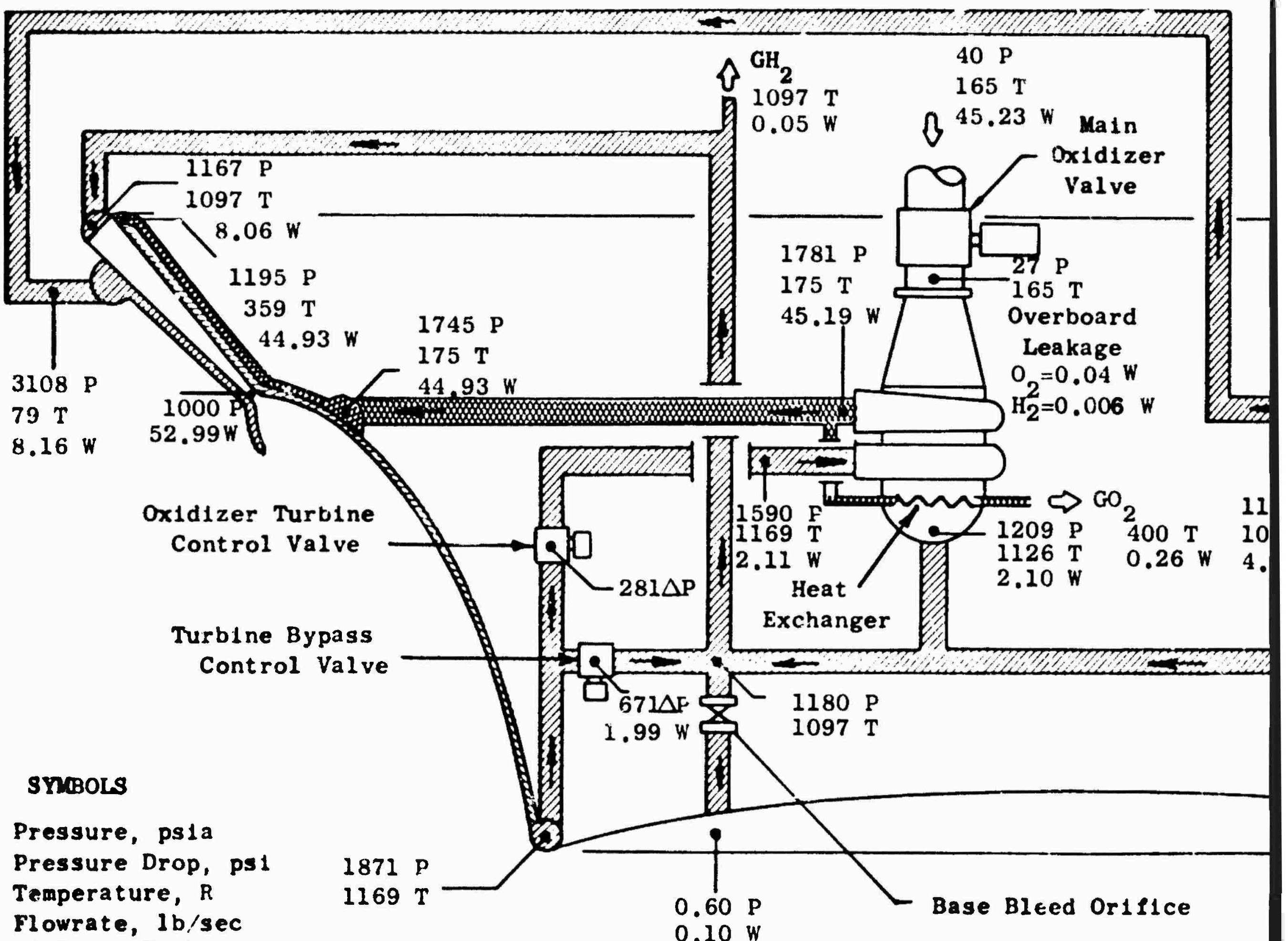
* $\Delta F_{BL}/\dot{M}_{Injector}$

** $F_{secondary}/\dot{M}_{secondary}$

TABLE 27. O_2/H_2 AEROSPIKE ENGINE OPERATING PARAMETERS, DOUBLE-PANEL COOLING CIRCUIT

Thrust, pounds	25,000	25,000	25,000	5000	5000	5000
Engine Mixture Ratio	5.0	5.5	6.0	5.0	5.5	6.0
Engine Specific Impulse, lb-sec/lbm	472.9	470.4	466.8	449.5	444.6	439.0
Nozzle Stagnation Pressure, psia	1013	1000	990	209	207	205
Total Oxidizer Flowrate, lb/sec	44.31	45.23	46.17	9.51	9.76	10.00
Total Fuel Flowrate, lb/sec	8.86	8.22	7.70	1.93	1.80	1.70
Oxidizer Turbine Flowrate, lb/sec	2.04	2.11	2.24	0.29	0.29	0.30
Fuel Turbine Flowrate, lb/sec	4.55	4.07	3.80	0.58	0.53	0.50
Turbine Bypass Flowrate, lb/sec	2.21	1.99	1.61	1.03	0.96	0.88
Base Flowrate, lb/sec	0.11	0.10	0.10	0.02	0.02	0.02
Base Pressure, psia	0.60	0.60	0.60	0.13	0.13	0.13
Thrust Chamber Mixture Ratio,	5.06	5.57	6.08	4.98	5.48	5.98
Oxidizer Injection Temperature, R	318	359	415	262	361	456
Fuel Injection Temperature, R	1068	1097	1097	1408	1454	1450
Turbine Inlet Temperature, R	1144	1169	1168	1451	1495	1491
Oxidizer Jacket Inlet Pressure, psia	1519	1745	2060	310	355	416
Oxidizer Injector Inlet Pressure, psia	1144	1195	1250	243	259	274
Oxidizer Jacket Pressure Loss, psi	375	550	810	68	96	143
Oxidizer Injector Pressure Loss, psi	126	190	255	33	51	68
Fuel Jacket Inlet Pressure, psia	3570	3108	2854	725	636	585
Fuel Injector Inlet Pressure, psia	1198	1167	1139	260	254	247
Fuel Jacket Pressure Loss, psi	1526	1237	1097	398	324	286
Fuel Injector Pressure Loss, psi	180	162	144	50	46	41
Oxidizer Pump Inlet Pressure, psia	27	27	26	39	39	39
Oxidizer Pump Discharge Pressure, psia	1554	1781	2097	312	357	418
Oxidizer Pump Speed, rpm	22420	23880	25780	9210	9950	10880
Oxidizer Pump Power, horsepower	369	434	528	18	22	28
Fuel Pump Inlet Pressure, psia	16	17	17	20	20	20
Fuel Pump Discharge Pressure, psia	3626	3156	2897	728	638	588
Fuel Pump Speed, rpm	80160	74980	71710	35110	32850	31510
Fuel Pump Power, hp	2789	2270	1955	146	119	104
Oxidizer Turbine Inlet Pressure, psia	1581	1781	1621	294	293	293
Oxidizer Turbine Pressure Ratio,	1.275	1.315	1.371	1.097	1.119	1.150
Fuel Turbine Inlet Pressure psia, psia	2041	1870	1756	327	312	299
Fuel Turbine Pressure Ratio	1.670	1.573	1.516	1.230	1.201	1.186
Turbine Bypass Valve Pressure Loss, psi	807	671	592	38	31	28
Oxidizer Turbine Valve Pressure Loss, psi	462	281	137	33	19	6

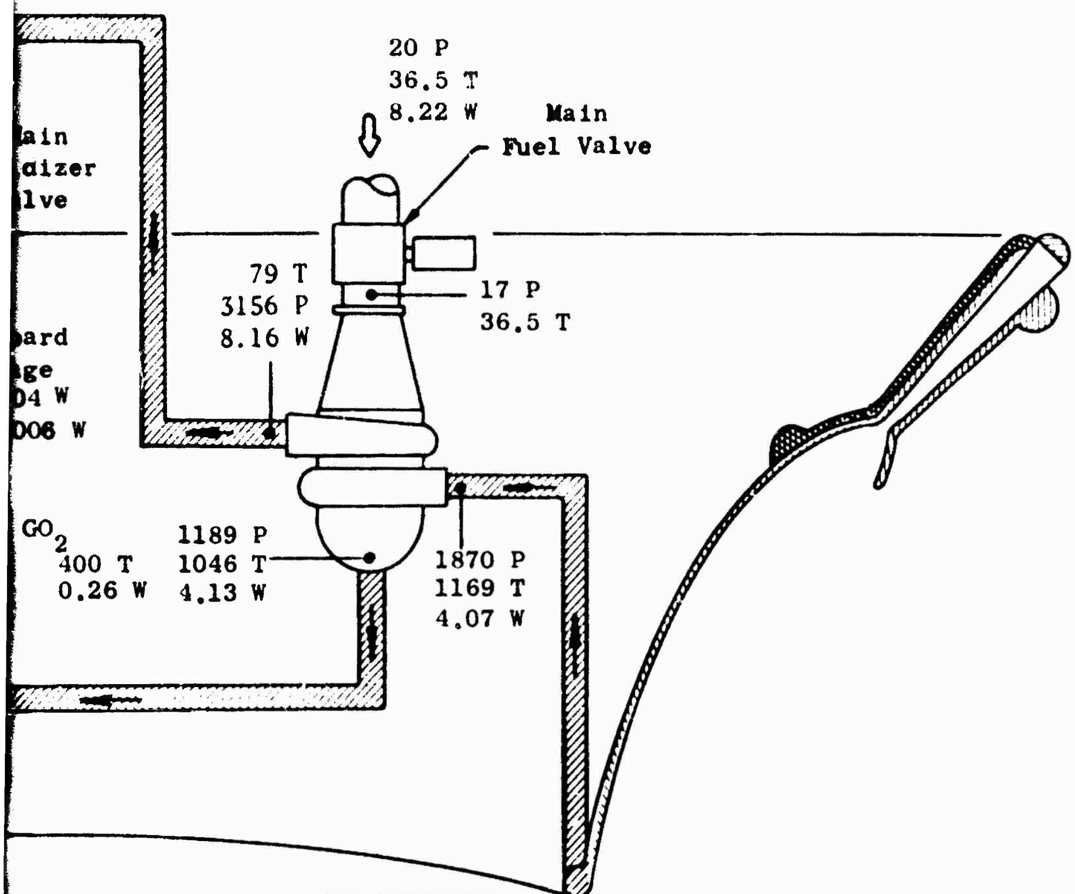
NOTE: Tank pressurization flows are included in engine mixture ratio, and excluded from engine specific impulse. Values are: 0.26 lb/sec gaseous oxygen, 0.05 lb/sec gaseous fuel.



SYMBOLS

P - Pressure, psia
ΔP - Pressure Drop, psi
T - Temperature, R
W - Flowrate, lb/sec
GH_2 and GO_2 - Tank Pressurization Flow, lb/sec

Thrust, lb	
Total	25,00
Primary	24,38
Secondary	62

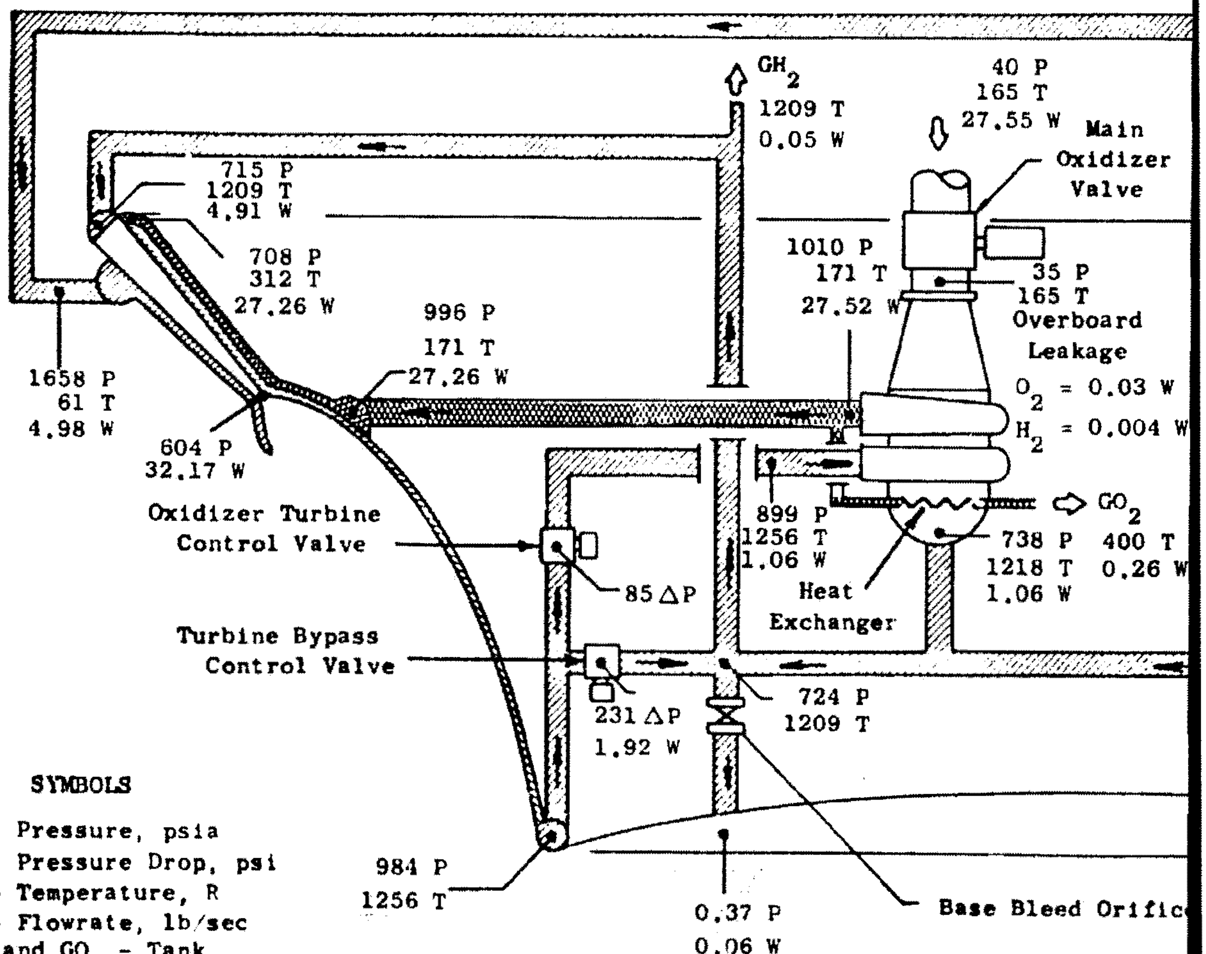


rust, lb	
tal	25,000
imary	24,380
condary	620

Mixture Ratio, o/f	
Pump Inlet	5.50
Primary	5.57

Figure 19. 25,000-Pound-Thrust, Double-Panel Aerospike Engine System Operating Conditions (25,000-Pound Operating Thrust)

2

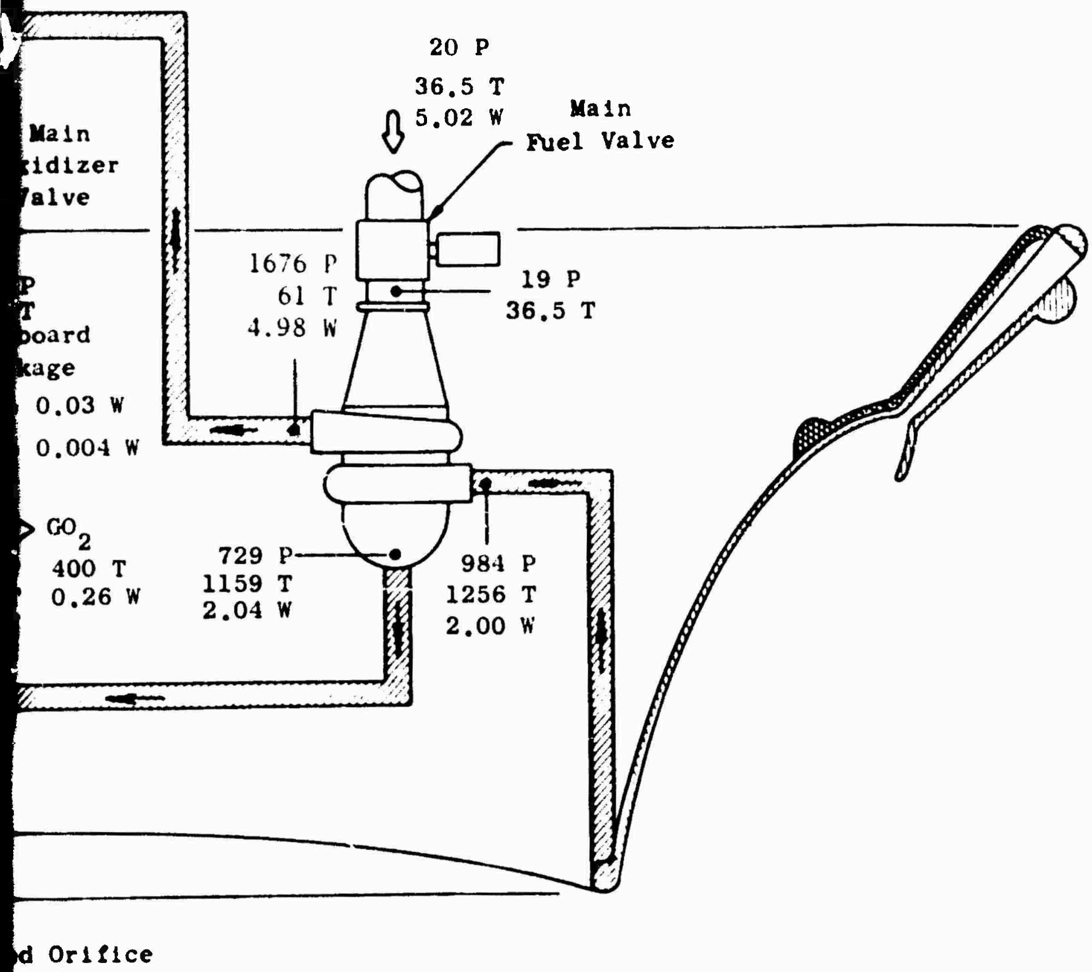


SYMBOLS

P - Pressure, psia
ΔP - Pressure Drop, psi
T - Temperature, R
W - Flowrate, lb/sec
GH_2 and GO_2 - Tank Pressurization Flow, lb/sec

Thrust, l

Total
Primary
Secondary

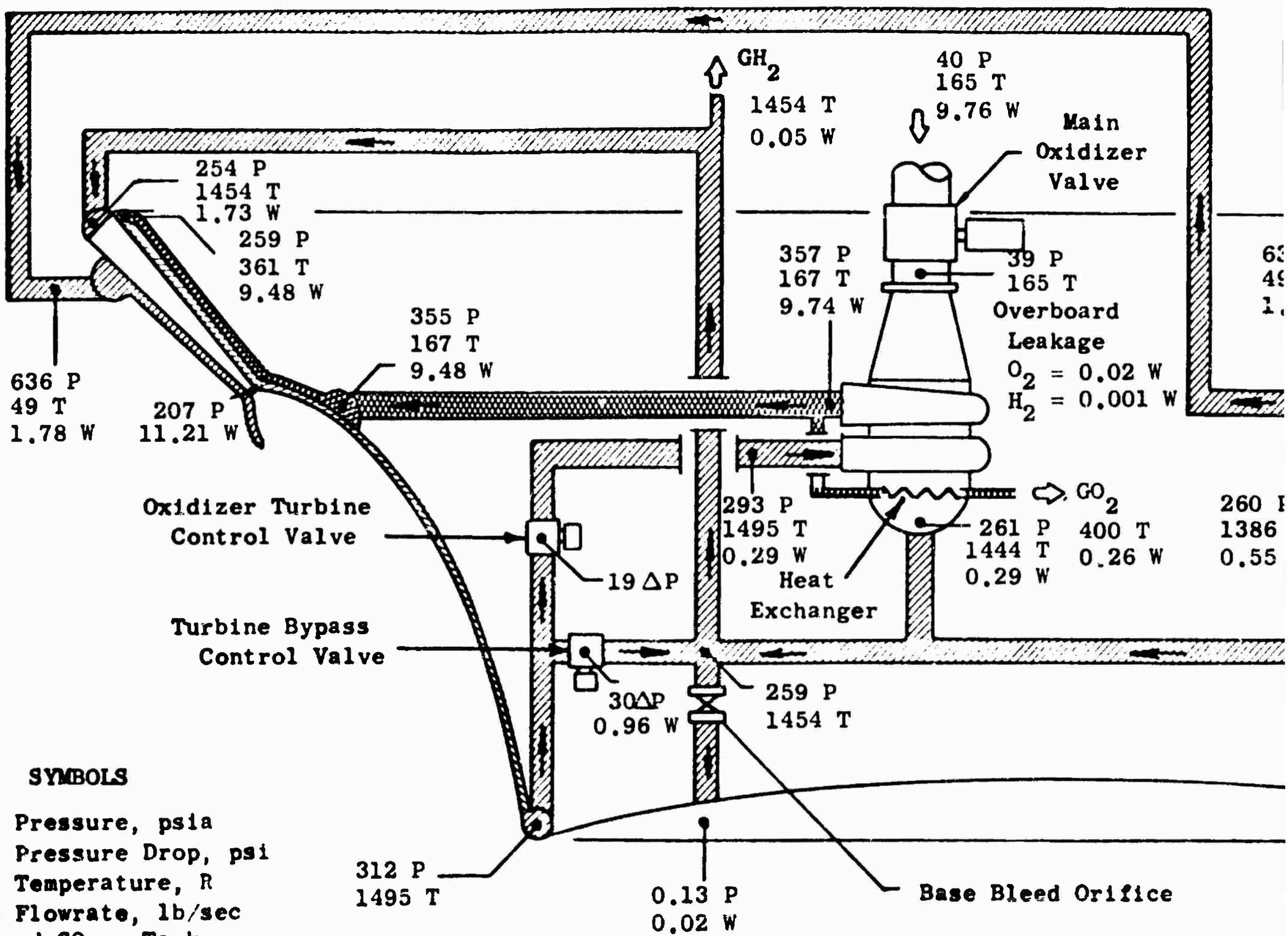


hrust, lb	
otal	15,000
rimary	14,620
econdary	380

Mixture Ratio, o/f	
Pump Inlet	5.50
Primary	5.55

Figure 20. 25,000-Pound-Thrust, Double-Panel Aerospike Engine System Operating Conditions (15,000-Pound Operating Thrust)

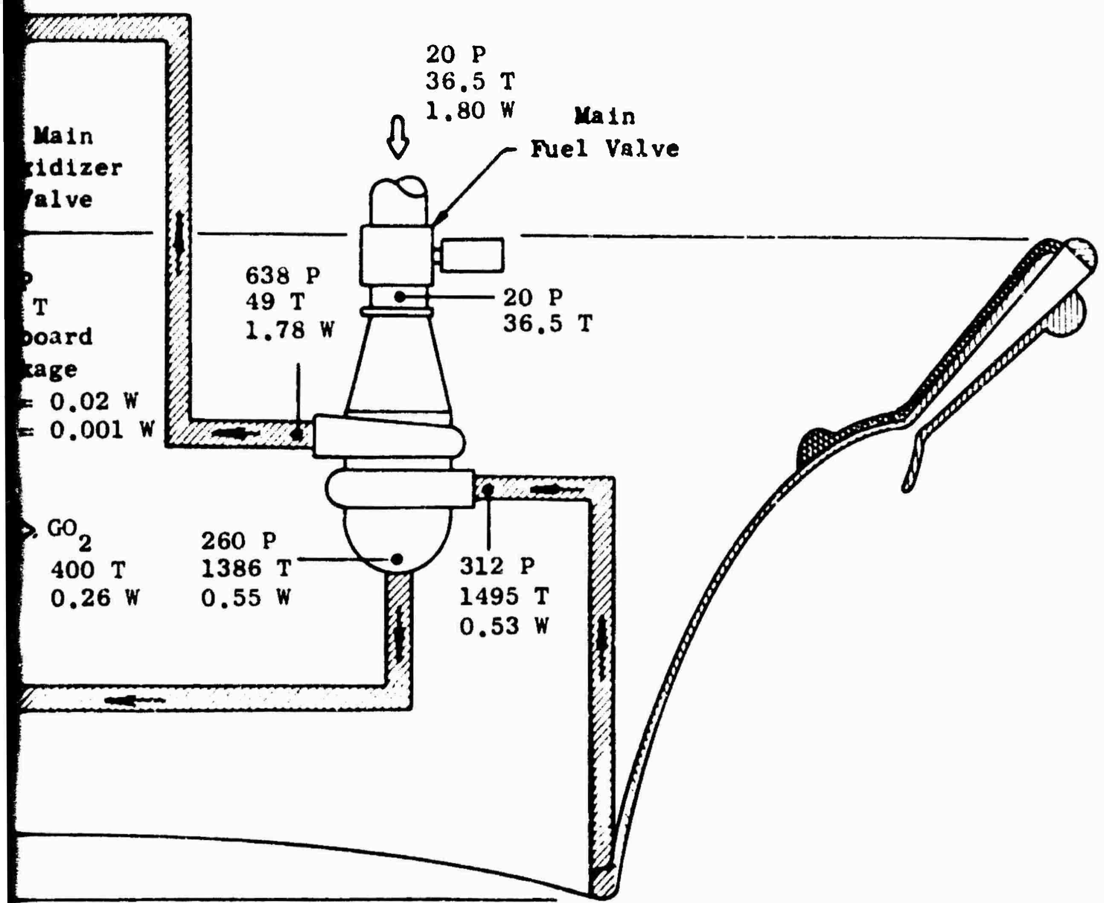
2



SYMBOLS

P - Pressure, psia
ΔP - Pressure Drop, psi
T - Temperature, R
W - Flowrate, lb/sec
GH$_2$ and GO$_2$ - Tank Pressurization Flow, lb/sec

Thrust, lb	
Total	5000
Primary	4870
Secondary	130



hrust, lb	
otal	5000
rimary	4870
econdary	130

Mixture Ratio, o/f	
Pump Inlet	5.50
Primary	5.48

Figure 21. 25,000-Pound-Thrust, Double-Panel Aerospike Engine System Operating Conditions (5000-Pound Operating Thrust)

2

Slight variations in engine operating conditions at the nominal design point are noted when compared with those presented in the component design description section. The design of the individual engine components and the engine system operational analysis are actually an iterative process where an initial nonlinear balance analysis is conducted with preliminary estimates of pump and turbine operating maps and component flow resistances. The individual component design studies are then conducted based on these initial balance results. Certain system operational characteristics and component design details are refined as a result of detailed studies. After the design analysis of the individual components and the final pump and turbine operating maps are completed, a final nonlinear engine balance analysis is completed. Slight variations in certain component operating conditions also will influence the operating characteristics of adjacent components in the system. For example, a slight variation in thrust chamber cooling jacket ΔP or bulk temperature rise will result in small variations in the nominal turbopump operating conditions. These variations are usually small and well within the design capability of the individual components. In some cases, minor changes to certain component design parameters are warranted before detail design to ensure maximum or optimum performance of that component.

The results of the nonlinear balance presented in this section are those obtained from this final analysis.

ENGINE POWER MARGIN

One attractive feature of the selected expander topping cycle is the available power margin for development contingencies. If, after the hardware has been manufactured, component ΔP values are higher than the anticipated nominal design values, or turbopump efficiency estimates are not met, additional pump power requirements can be met by using some of the nominal turbine bypass flow. The influence of several important parameter variations on engine operation is shown in Table 28.

TABLE 28. 25,000-POUND-THRUST O_2/H_2 A

POWER MARGIN AVAILABILITY

	Hydrogen ΔP, Pump Discharge To Fuel Turbine Inlet	Hydrogen ΔP, Fuel Turbine Discharge To Chamber Pressure	Turbine Inlet Temperat
Current Nominal Parameter Value At F=25,000 lb, MR=5.5:1	1286 psi	184 psia	1169 R
Operating p it with least nominal engine margin			
Thrust poinds	25,000	25,000	25,000
Mixture Ratio	5.0:1	6.0:1	6.0:1
Parameter value at this thrust and MR	1585 psi	165 psi	1168 R
Operating Characterictics for allowable value of parameter			
Allowable Parameter Value	2417 psi (+52%)	519 psi (+214%)	827 R -10%)
Turbine Bypass Flow, Percent of total fuel	20.2	5.0 **	5.0
Oxidizer Turbine Control Value ΔP, psi	600	134	205
Fuel Pump Speed, rpm	88,000	75,860	72,560

*Pump head coefficient reduced simultaneously with efficiency, at one-half of efficien

**Underlined values are limits

/

OUND-THRUST O_2/H_2 AEROSPIKE ENGINE

MARGIN AVAILABILITY

Fuel harge r	Turbine Inlet Temperature	Turbomachinery Performance			
		Oxidizer Turbine Efficienty	Fuel Turbine Efficiency	Both Turbine Efficiency	Fuel* Pump Performance
	1169 R	0.445	0.756	0.445/0.756	0.620
	25,000	10,000	25,000	25,000	25,000
	6.0:1	6.0:1	6.0:1	6.0:1	5.0:1
	1168 R	0.373	6.755	0.447/0.755	0.619
	827 R -10%)	0.335 (-40%)	0.456 (-32%)	0.301/0.519 (-32%)	0.463 (-25%)
	<u>5.0</u>	42.3	<u>5.0</u>	<u>5.0</u>	15.8
	205	<u>6.5</u>	540	250	780
	72,560	41,950	75,560	74,400	88,000

one-half of efficiency reduction.

2

Allowable engine system or component variations, which result in either a minimum 5-percent turbine bypass, a minimum control valve pressure drop, or a maximum 88,000-rpm fuel pump speed, are presented. In all instances but one, thrust level is 25,000 pounds; however, power limits (minimum bypass flow) are encountered usually at a mixture ratio of 6.0, whereas speed limits are encountered at a mixture ratio of 5.0. Turbopump performance, component pressure drops, and turbine inlet temperature variations were considered. These variations can be accomodated through engine control adjustments without hardware change or with a change of the turbine nozzle block. A brief discussion regarding selection of the power margin parameters follows. The limiting parameter is underlined for each case in Table 28.

Minimum turbine bypass flow was selected arbitrarily as 5 percent of total fuel flow, compared to the nominal value of 24 percent. This still allows adequate bypass flow for control purposes. The oxidizer turbine control valve has a smaller flow area than the turbine bypass valve. Examination of oxidizer turbine and/or pump performance degradation required that the oxidizer turbine control valve area be increased, so a maximum allowable area was selected equal to the maximum area required of the turbine bypass control valve, which implies a substitution of the larger flow area valve if additional oxidizer turbine flow is required. The available power limit occurs at 10,000 pounds thrust and results in an oxidizer turbine control valve ΔP of 6.5 psi. An alternative to increasing valve area would be a slight reduction in fuel turbine admission area, which has the effect of increasing fuel turbine pressure ratio to increase the oxidizer turbine control valve nominal pressure drop which is in parallel with the fuel turbine. A slight reduction in nominal balance turbine bypass flow would result. The third limit parameter is fuel pump speed. The design upper operating limit was selected early in the program as 80,000 rpm. This value is encountered, however, at 25,000 pounds thrust and a mixture ratio of 5.0 for the nominal engine. Because no margin then exists, pump design will have to be altered to allow a reasonable margin. A 10-percent increase was selected, resulting in a limit value of 88,000 rpm.

Double-Panel Engine Transient Operation

Engine start and cutoff operation and control sequencing were investigated with the use of a computerized dynamic model of the engine system. The computer model is a set of nonlinear differential equations describing the significant physical processes affecting engine transient operation. All major components are represented in terms of operating maps, mechanical and fluid inertias, priming volumes, variable fluid properties, and heat transfer. The model has descriptions of heat transfer in both pumps, the fuel high-pressure duct, and the thrust chamber cooling circuits. A start and cutoff sequence and procedure was established which results in safe control of the engine system and component operating conditions during the transients.

Engine Start. The engine start transients were based on tank pressures described as follows:

	Oxygen	Hydrogen
Tank Pressure, psia	40	20
Line and Valve Pressure Drop at Full Thrust, psi	15	5

Initial hardware temperatures of 400 R were used. Both tank pressure and initial engine temperature will vary during a mission and the control system was established to allow for these variations.

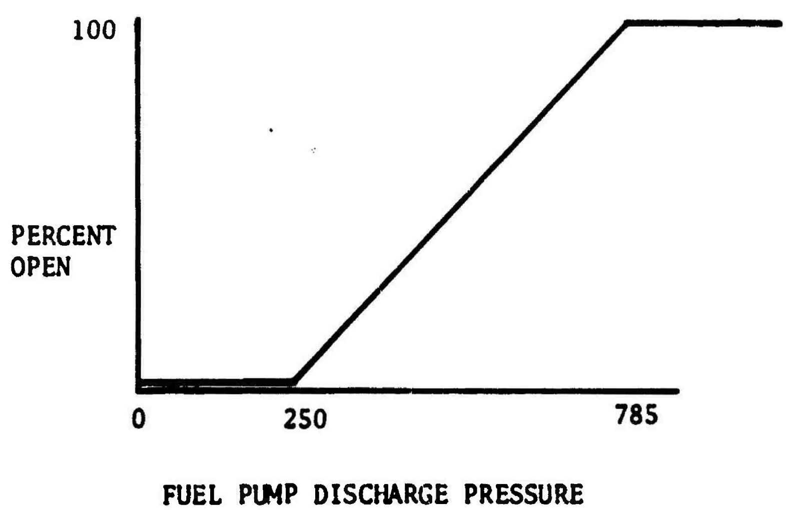
The initial part of the engine start is controlled in an open-loop fashion. As thrust begins to build up, closed-loop control is initiated. There are two reasons for incorporating closed-loop control into the engine start sequence.

First, the engine has a 5:1 throttling requirement and, therefore, starting to various thrust levels will be required. This can best be achieved by using a closed-loop control system. Second, a closed-loop control system can be used to minimize the effects of variations in start conditions (tank pressures, hardware temperatures, etc.) on the engine start transients. The two control valves (turbine bypass and oxidizer turbine inlet) both have proportional plus integral control. Thus, the command valve position can be expressed as:

$$\text{Position} = K_{1*} \text{ Error} + K_2 \int \text{Error } dt$$

The proportional part of the turbine bypass valve was made a function of command chamber pressure to compensate for variations in engine response at various thrust levels. No cross-compensation was used between the two control valves. That is, the only input to the turbine bypass valve was chamber pressure error, and the only input to the oxidizer turbine inlet valve was mixture ratio error.

The engine valve sequencing for start is shown in Fig. 22. Open-loop operation is used for the first 3 seconds with the valves scheduled as indicated. To prevent high fuel pump discharge pressure and pump speed during restart (when the pumps are chilled), the turbine bypass valve was made a function of fuel pump discharge pressure. The relationship used prior to 3.0 seconds was:



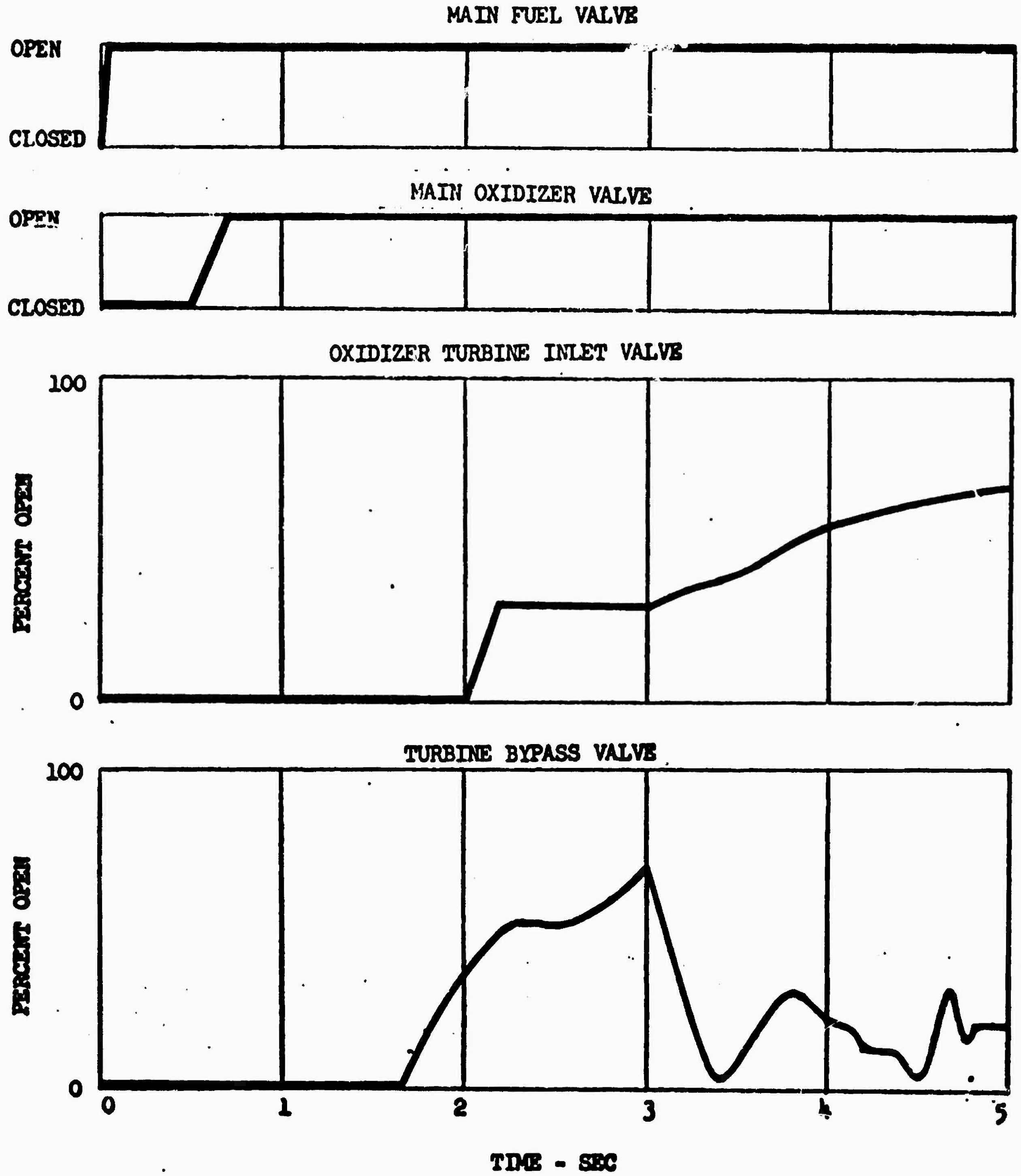


Figure 22, Engine Start Sequence For Ambient Start With 20-psia Fuel Tank Pressure

Closed-loop control is used after 3 seconds of operation. The command chamber pressure was 200 psia until 3.0 seconds, and then ramped to 1000 psia in 1.5 seconds. The mixture ratio command was 4.5 until 3.0 seconds, and then was ramped to 5.5 at 4.5 seconds. With this sequence, the engine achieves full thrust in less than 5 seconds as indicated in Fig. 23 .

In the start, the main fuel valve opens at engine start signal. With the oxidizer turbine inlet and the turbine bypass valves closed, initial power to the fuel turbine is provided while allowing the hydrogen flow (Fig. 24) to begin cooling the fuel pump, high-pressure duct, and thrust chamber. The heat transfer in the fuel pump reduces the density of the hydrogen in the pump, thereby reducing the fuel pump outlet pressure at any given pump speed.

The main oxidizer valve is opened between 0.5 and 0.7 speed after start signal. Between 0.5 and about 1.25 seconds, the oxidizer flow is priming the volume between the inlet valve and the oxidizer panel and chilling the oxidizer pump. By 1.25 seconds, adequate fuel pump speed has been obtained to prevent high chamber mixture ratio when the oxidizer side is primed. Figure 25 shows a thrust chamber mixture about 2.5 at 1.25 seconds. Two mixture ratios are shown in Fig. 25 . The engine mixture ratio is based on pump flowrates and the thrust chamber mixture ratio is based on injector flowrates. The primary difference is caused by the oxidizer priming. Initially, the oxidizer pump flow is high with only small amounts of oxidizer injector flow. Therefore, between 0.5 and 1.25 seconds, the thrust chamber is operating at low mixture ratios.

At 1.6 seconds, the turbine bypass is opened. At 2.0 seconds, the oxidizer turbine valve is opened to 30 percent of full open. These two positions are intended to produce a thrust chamber pressure of 200 to 300 psia. This control facilitates starting the engine to various levels without having any substantial thrust chamber pressure overshoot. At 3.0 seconds, the engine operation is turned over to closed-loop control.

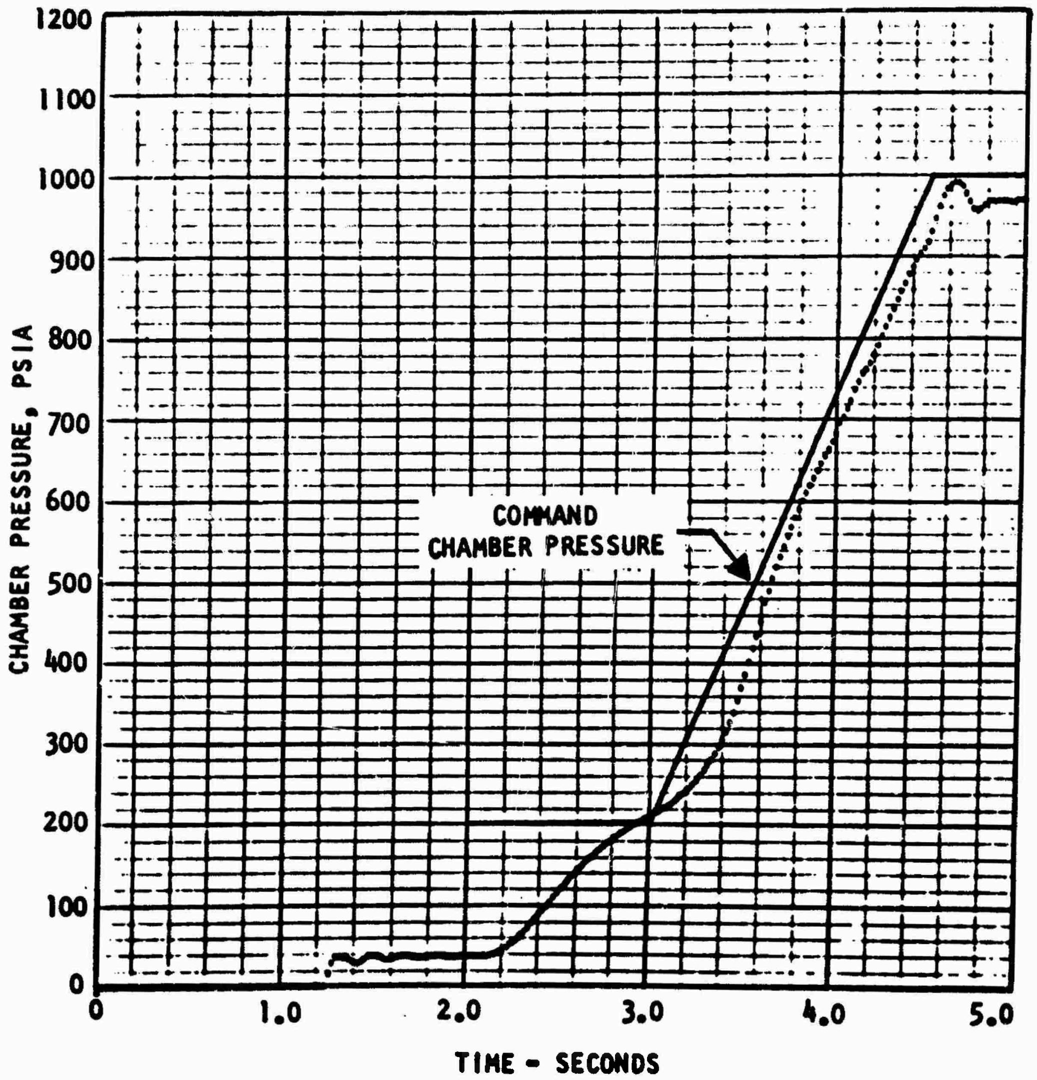


Figure 23. Thrust Chamber Pressure Versus Time For Ambient Start With 20-psia Fuel Tank Pressure

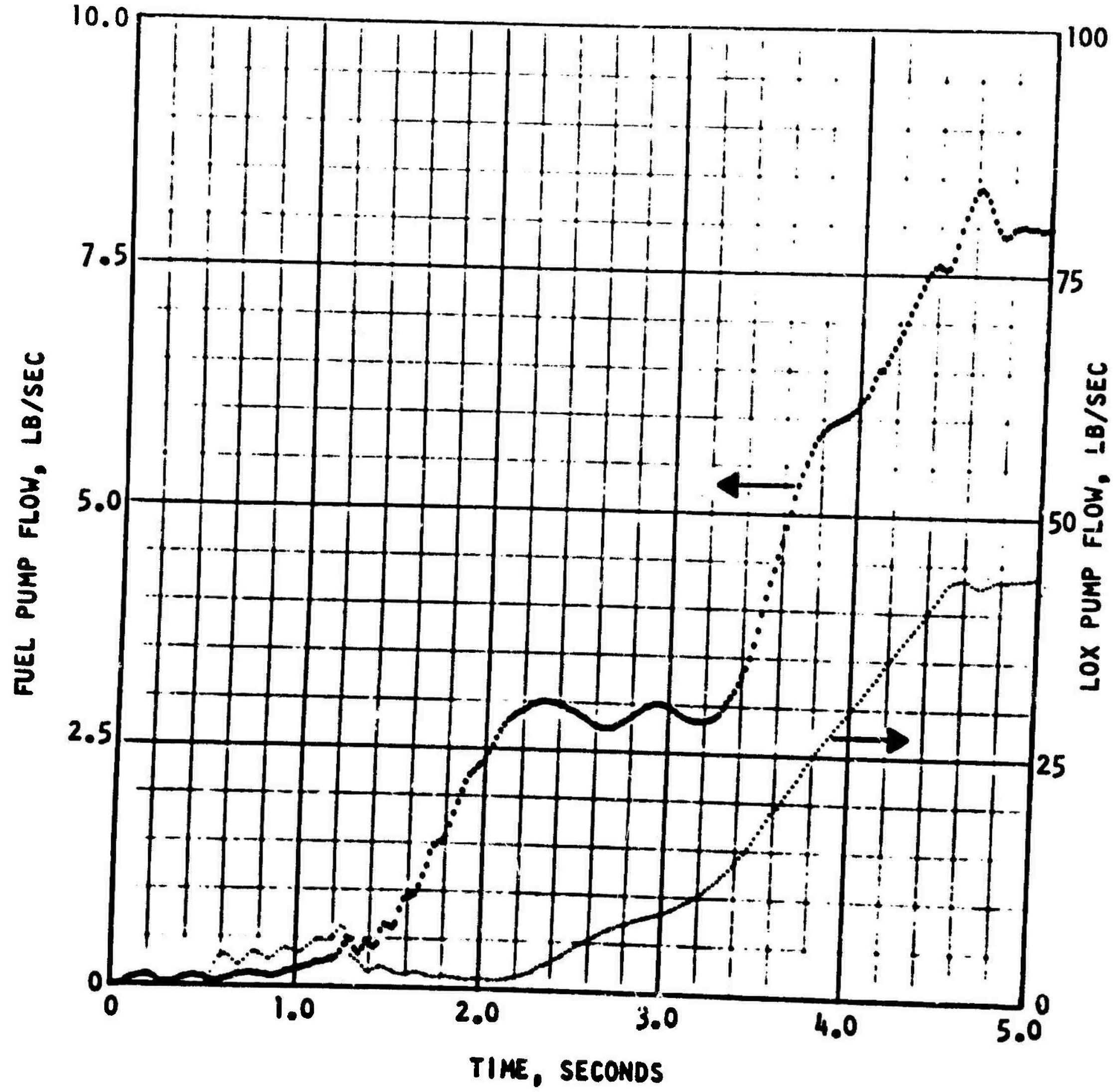


Figure 24. Pump Flowrates Versus Time For Ambient Start With 20-psia Fuel Tank Pressure

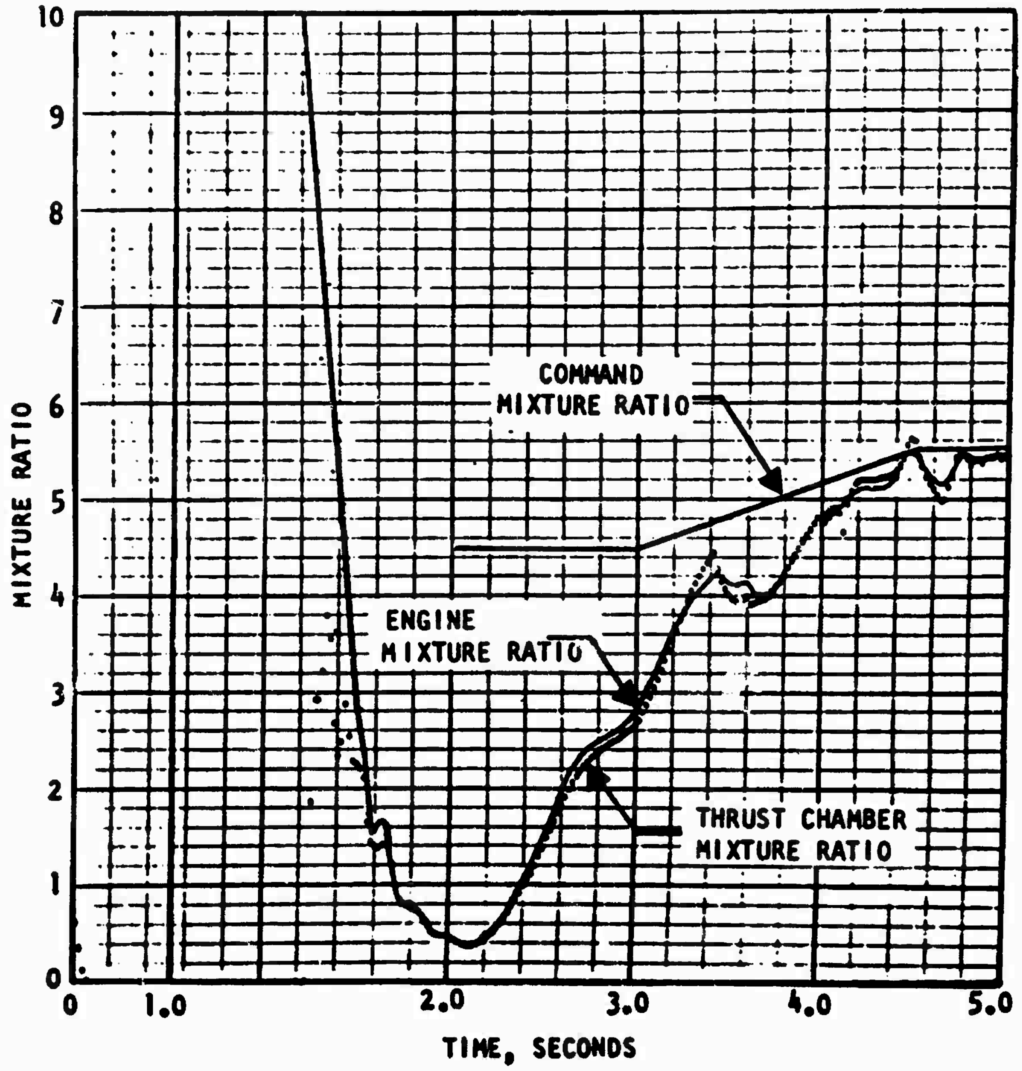


Figure 25. Mixture Ratio Versus Time for Ambient Start With 20-psia Fuel Tank Pressure

Pump speeds during the complete sequence are shown in Fig. 26. Valves shown are close to the steady-state design values and impose no design problem. The slight overshoot in fuel pump speed can be eliminated by optimizing the closed-loop control.

Alternate Initial Conditions. Engine operation using the described start sequence was investigated for immediate restart (pumps cold, turbine and thrust chamber warm) and at a 20-psia fuel tank pressure. Because of the turbine bypass control method, engine transients under these conditions were very similar to the nominal. The engine achieved full thrust in less than 5 seconds in all cases. Transients for the immediate restart are shown in Fig. 27 through 31.

Engine Cutoff. The engine cutoff is accomplished by the open-loop sequence shown in Fig. 32. Engine cutoff occurs in less than 4 seconds. To reduce pump speed in the final phases of shutdown, when bearing coolant may be irregular, the engine is essentially throttled to a low thrust and then cut off.

In engine shutdown, the turbine bypass valve is ramped to 50-percent open in 1.0 second and the oxidizer turbine inlet valve is closed to 25-percent open in 0.2 second. Because the fuel turbopump time constant is significantly lower than the oxidizer turbopump, closing the oxidizer turbine valve to 25 percent before opening the bypass valve maintains chamber mixture ratio at an acceptable value.

In 2.0 seconds, chamber pressure decreased to 140 psia (Fig. 33). At this time, the main oxidizer valve closure is initiated. At 2.25 seconds, the oxidizer system purge is turned on between the main oxidizer valve and the pump. By 2.5 seconds, the oxidizer has been purged from the pump and chamber pressure drops to 45 psia. To prevent fuel pump speed increase at this point, the turbine bypass valve is ramped to full open between 2.25 and 2.45 seconds. When the oxidizer is completely purged from the system (3.55 seconds), the main fuel valve is closed. Other engine characteristics are shown in Fig. 34 through 36.

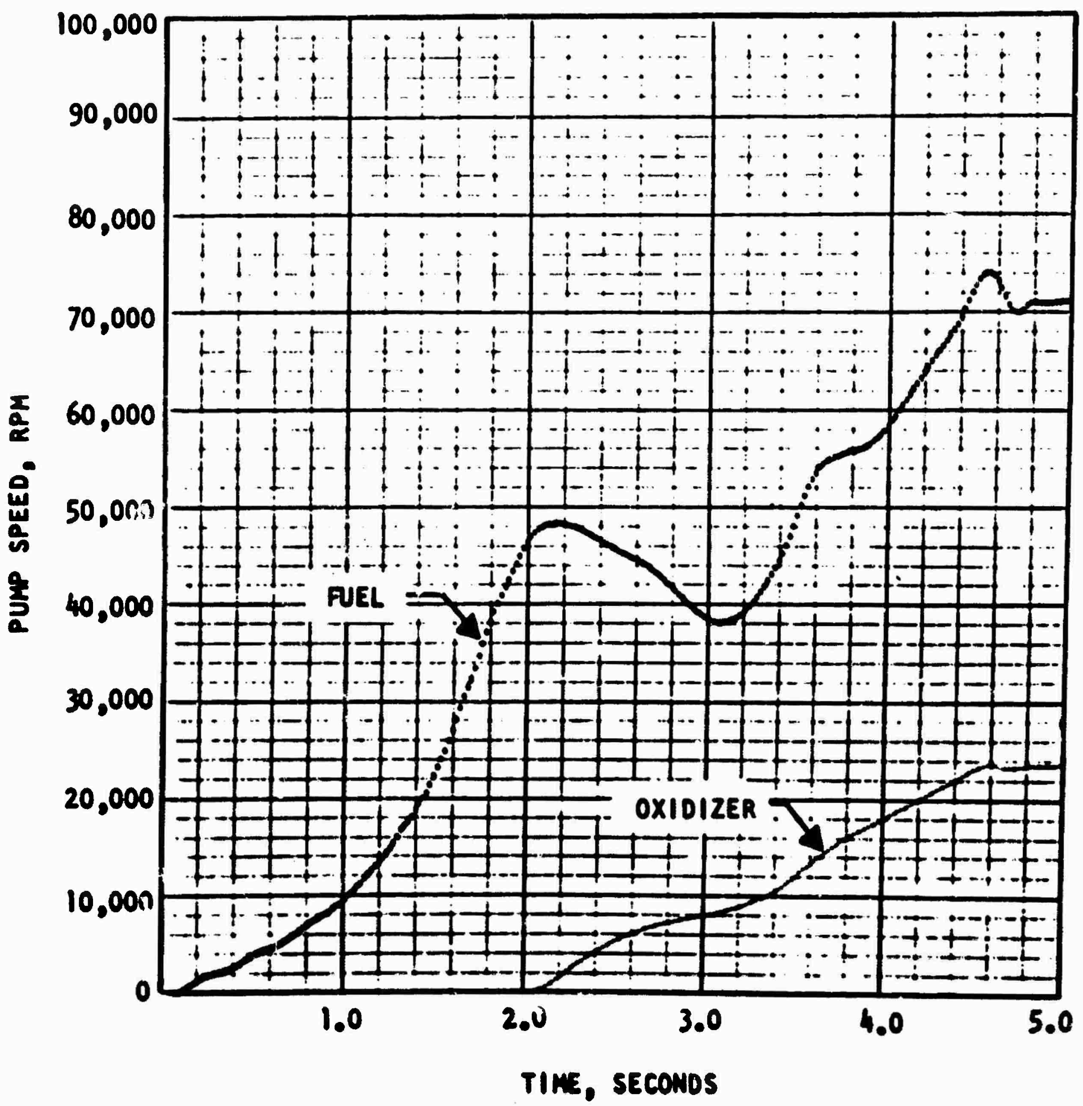


Figure 26. Pump Speeds Versus Time For Ambient Start With 20-psia Fuel Tank Pressure

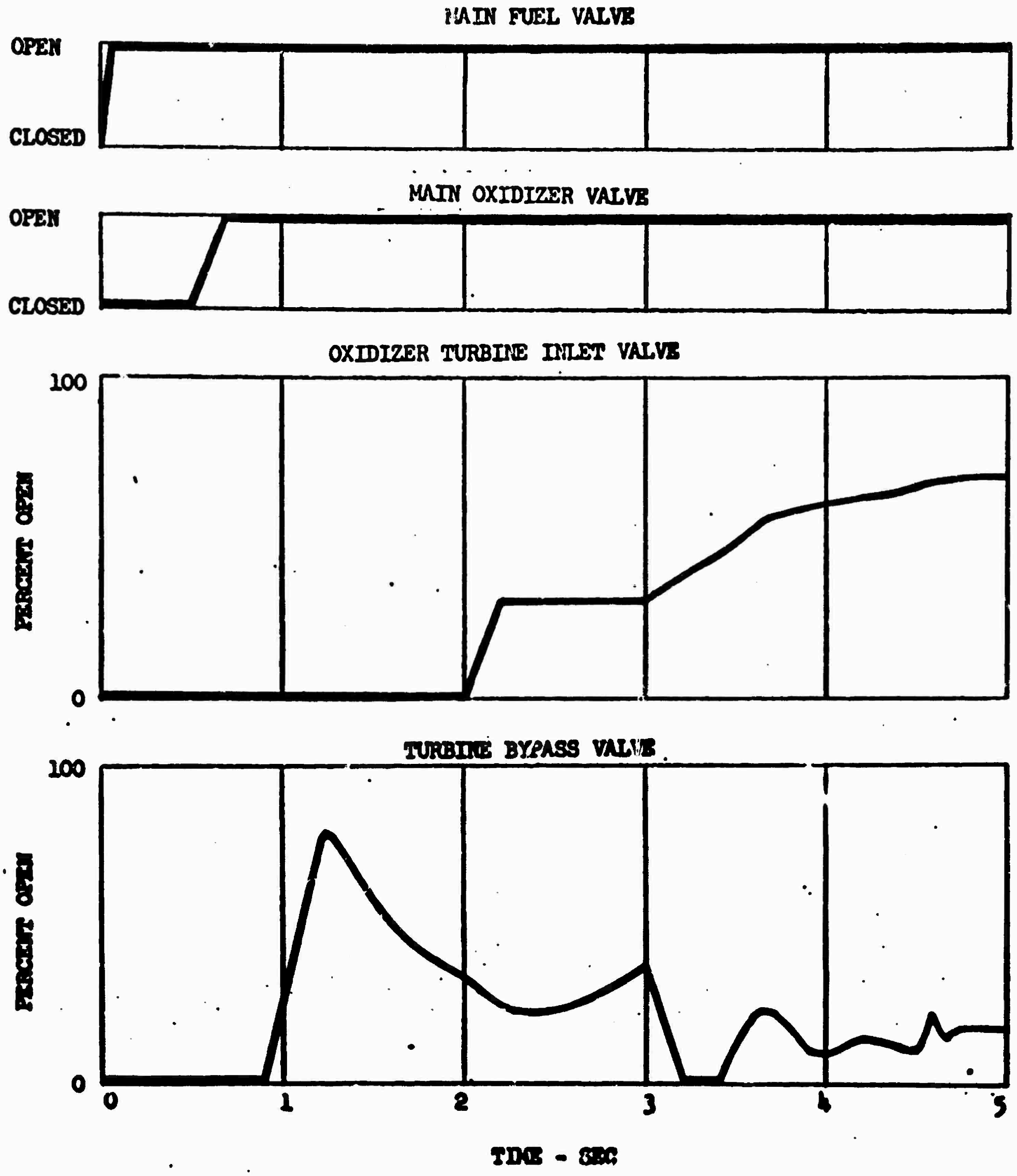


Figure 27. Engine Start Sequence For Immediate Restart With 20-psia Fuel Tank Pressure

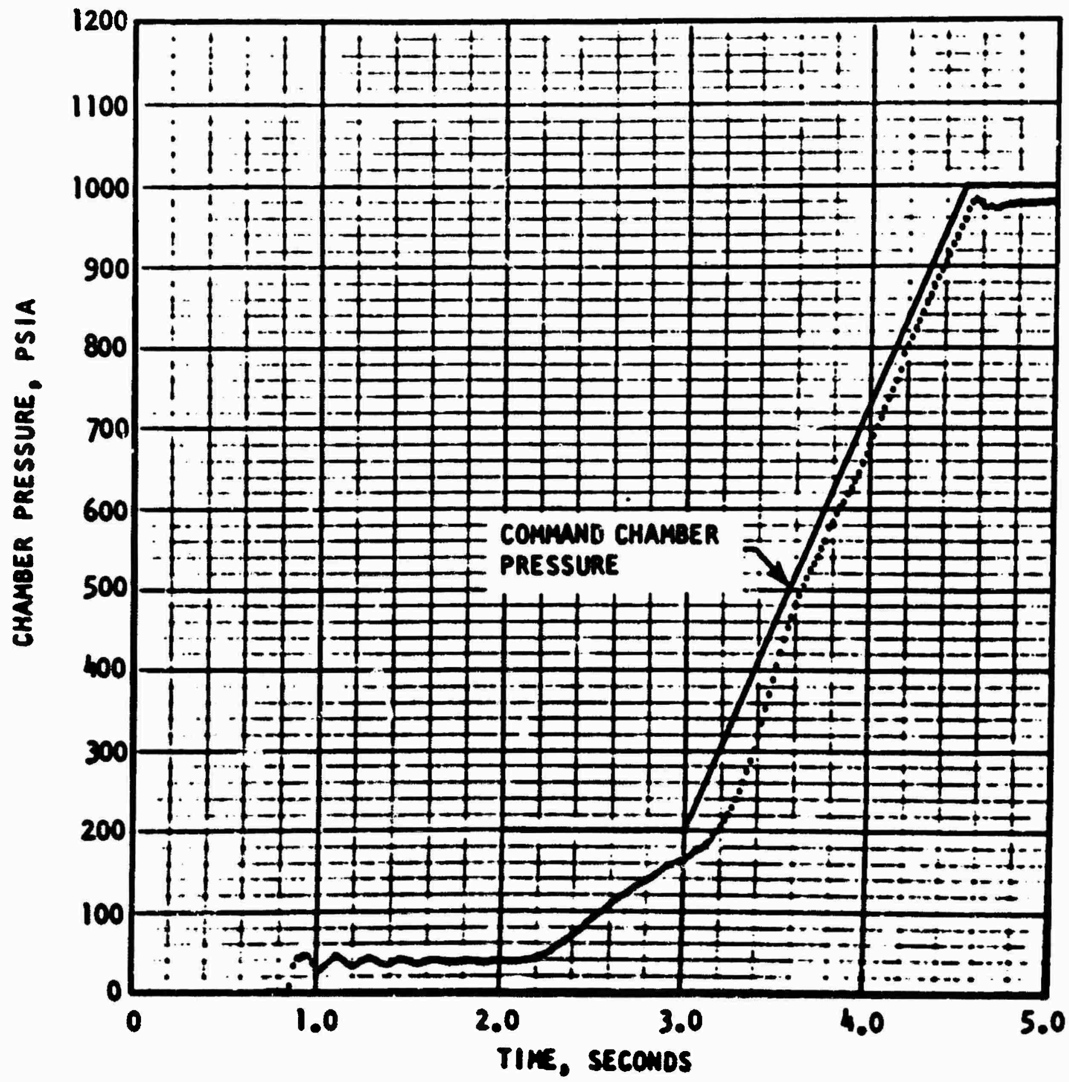


Figure 28. Thrust Chamber Pressure Versus Time For Immediate Restart With 20-psia Fuel Tank Pressure

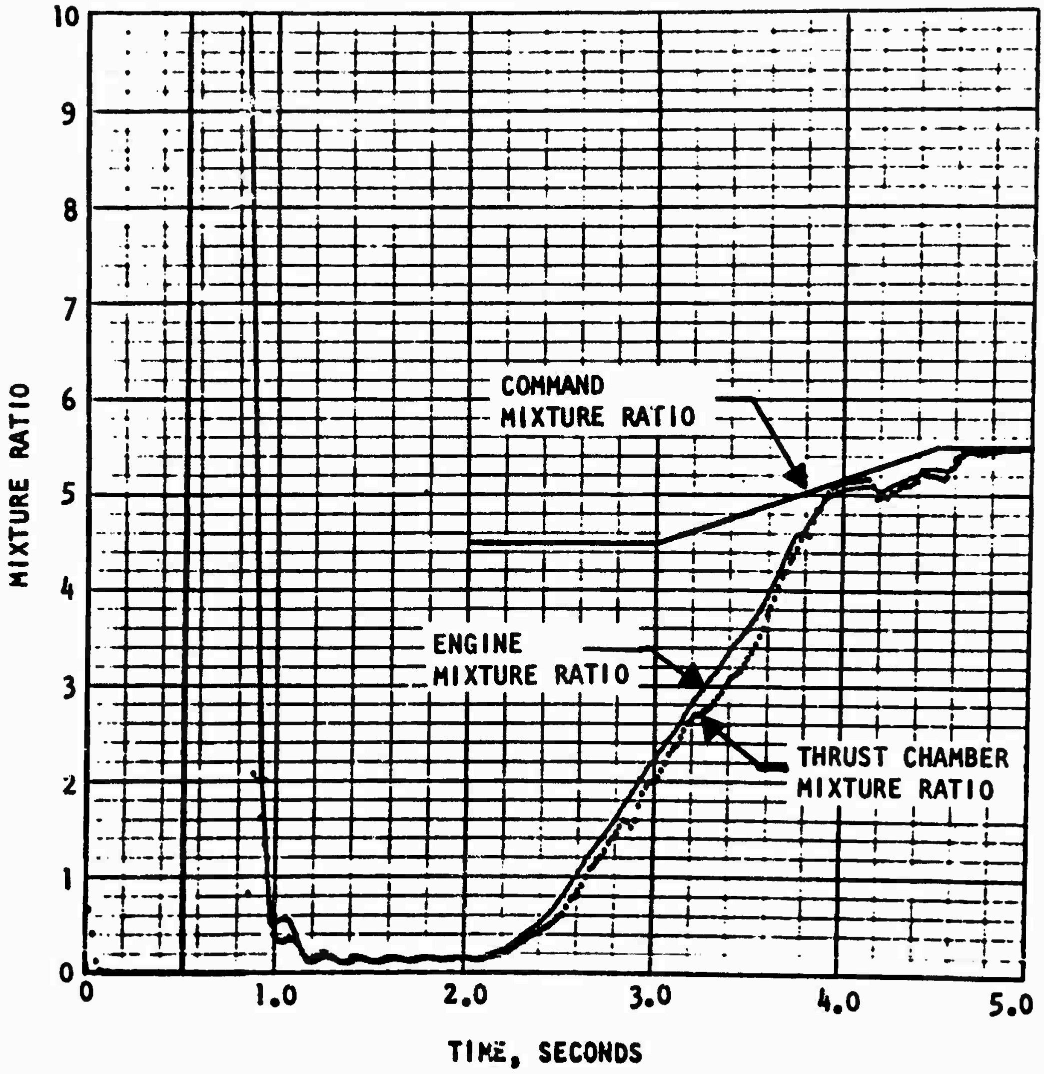


Figure 29. Mixture Ratio Versus Time for Immediate Restart With 20-psia Fuel Tank Pressure

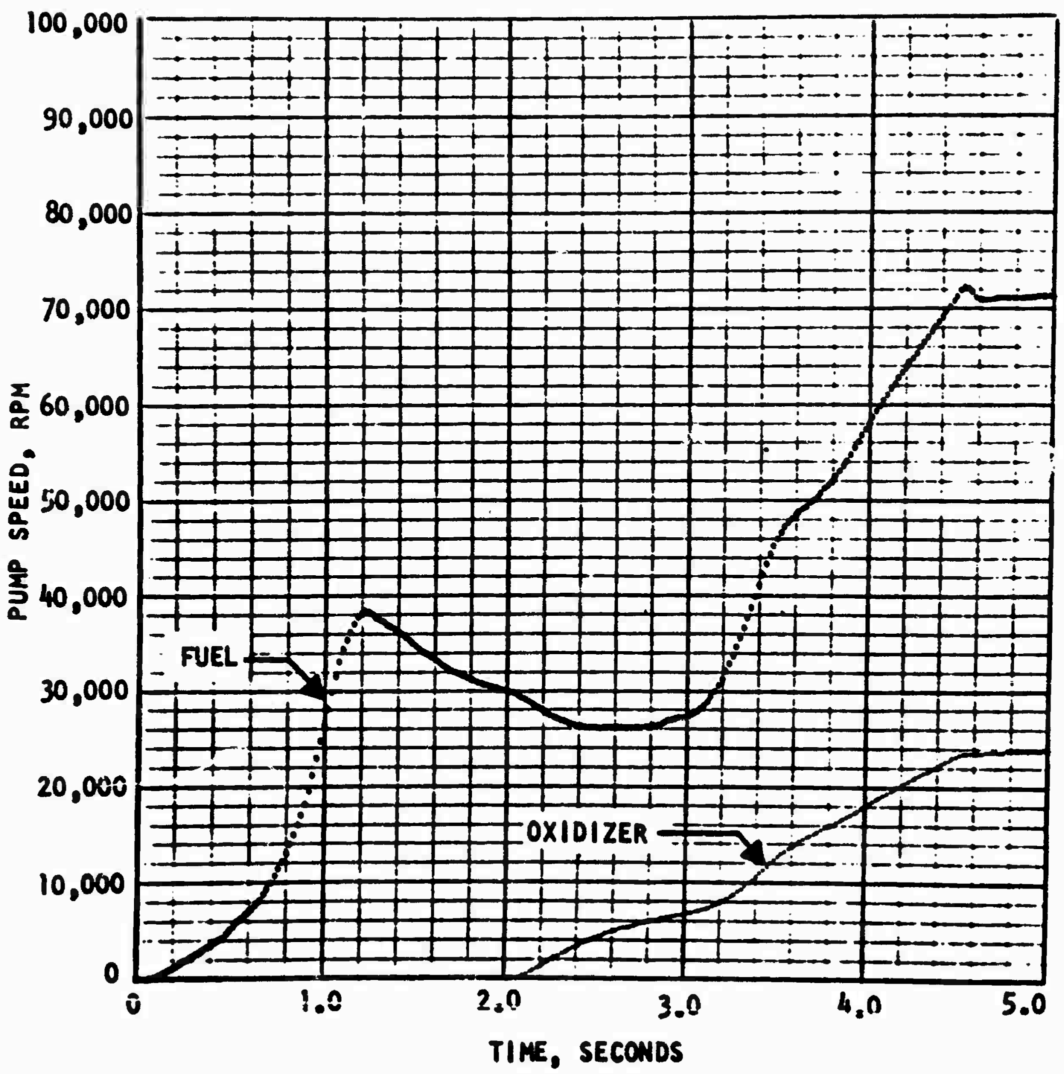


Figure 30. Pump Speeds Versus Time For Immediate Restart With 20-psia Fuel Tank Pressure

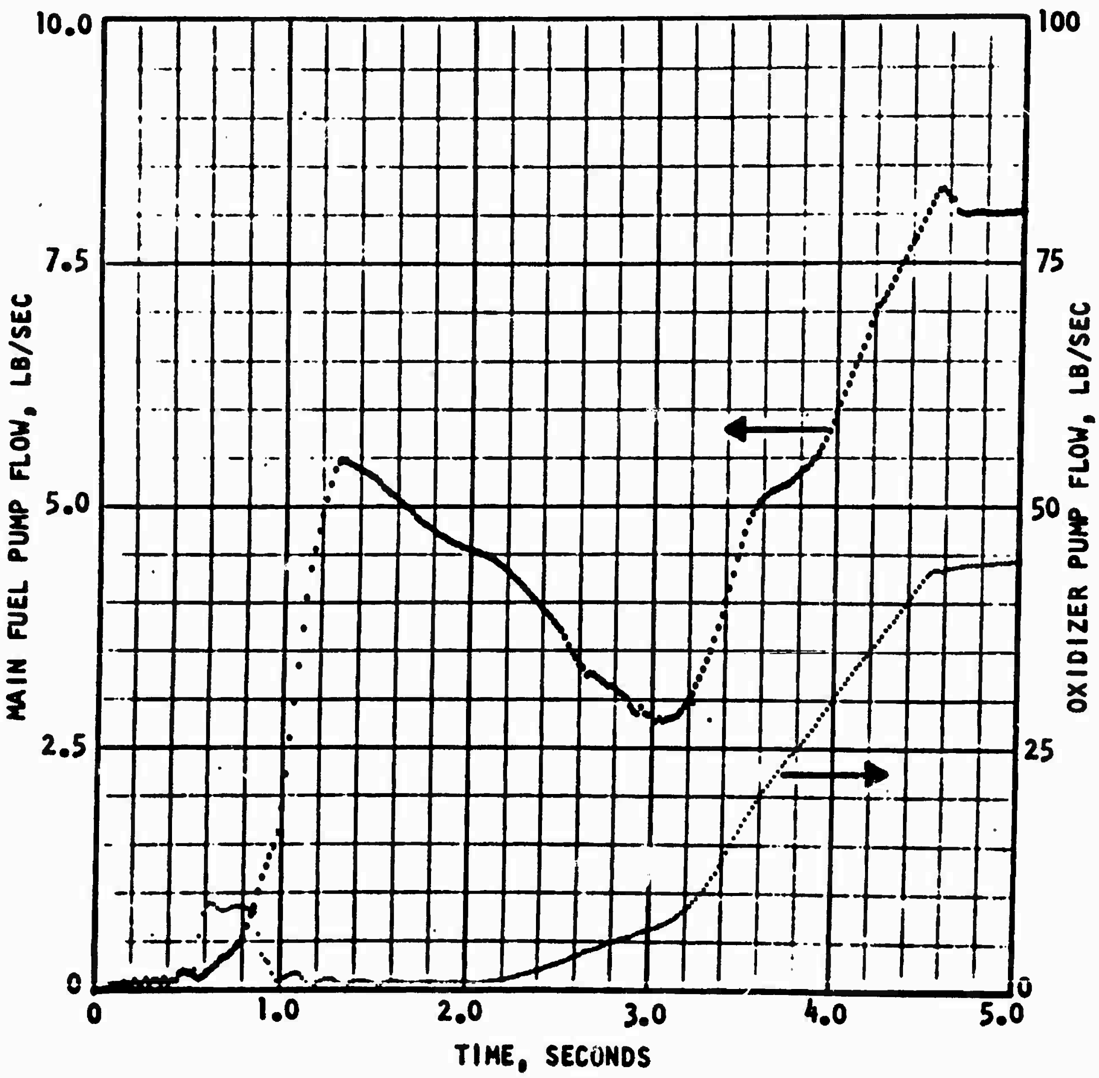


Figure 31. Pump Flowrates Versus Time For Immediate Restart With 20-psia Fuel Tank Pressure

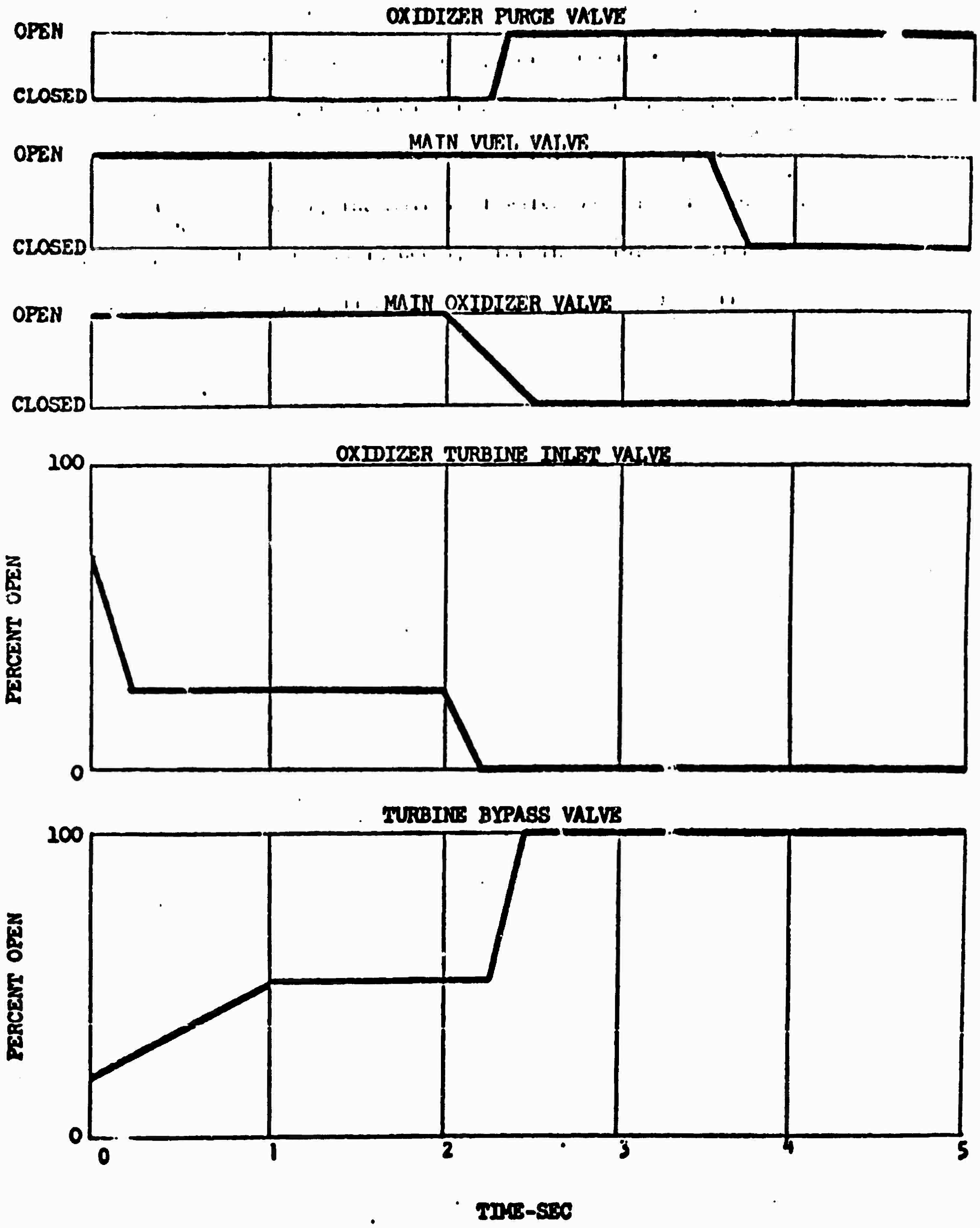


Figure 32. Engine Cutoff Sequence

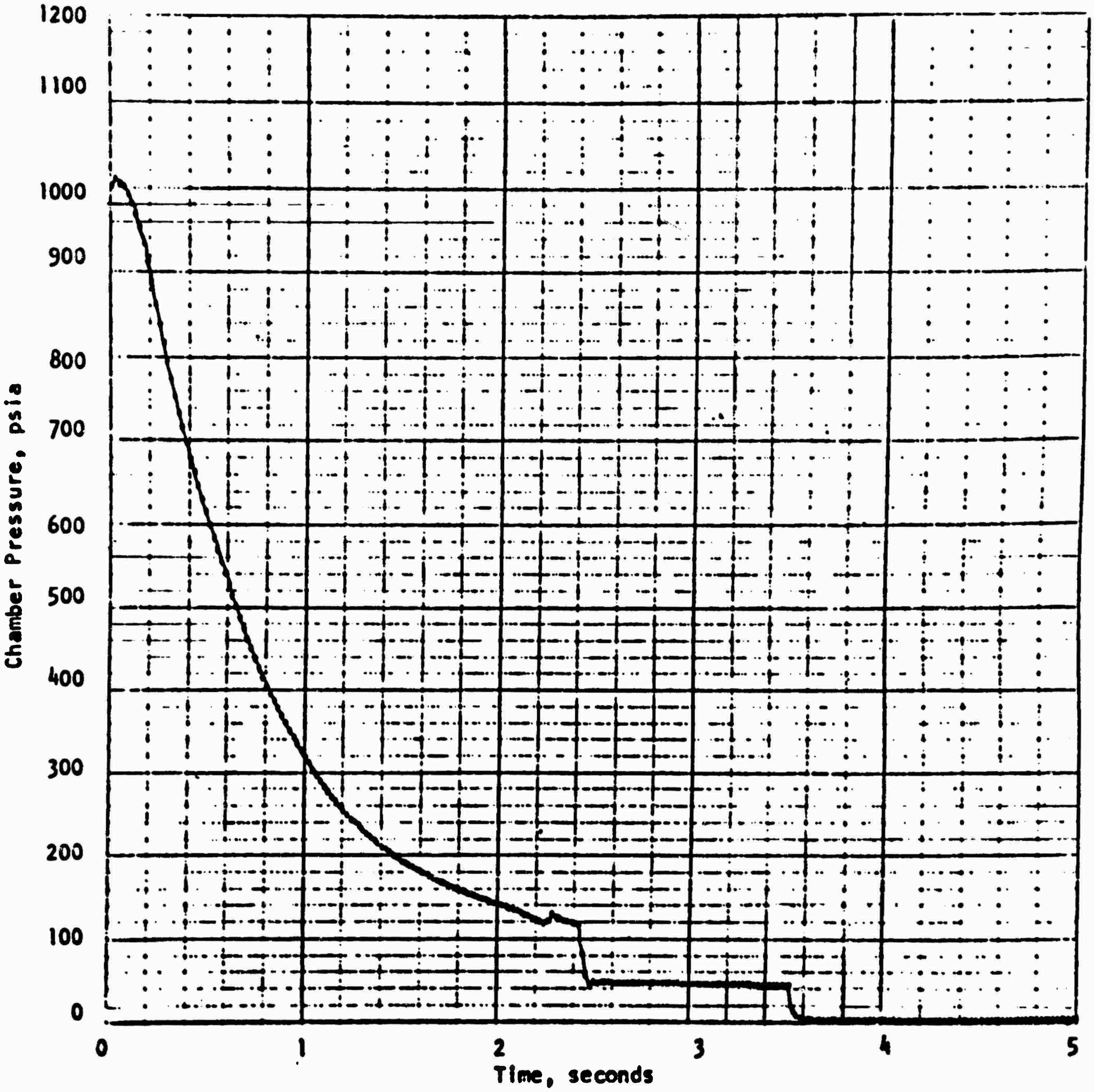


Figure 33. Thrust Chamber Pressure vs Time for Cutoff Simulation

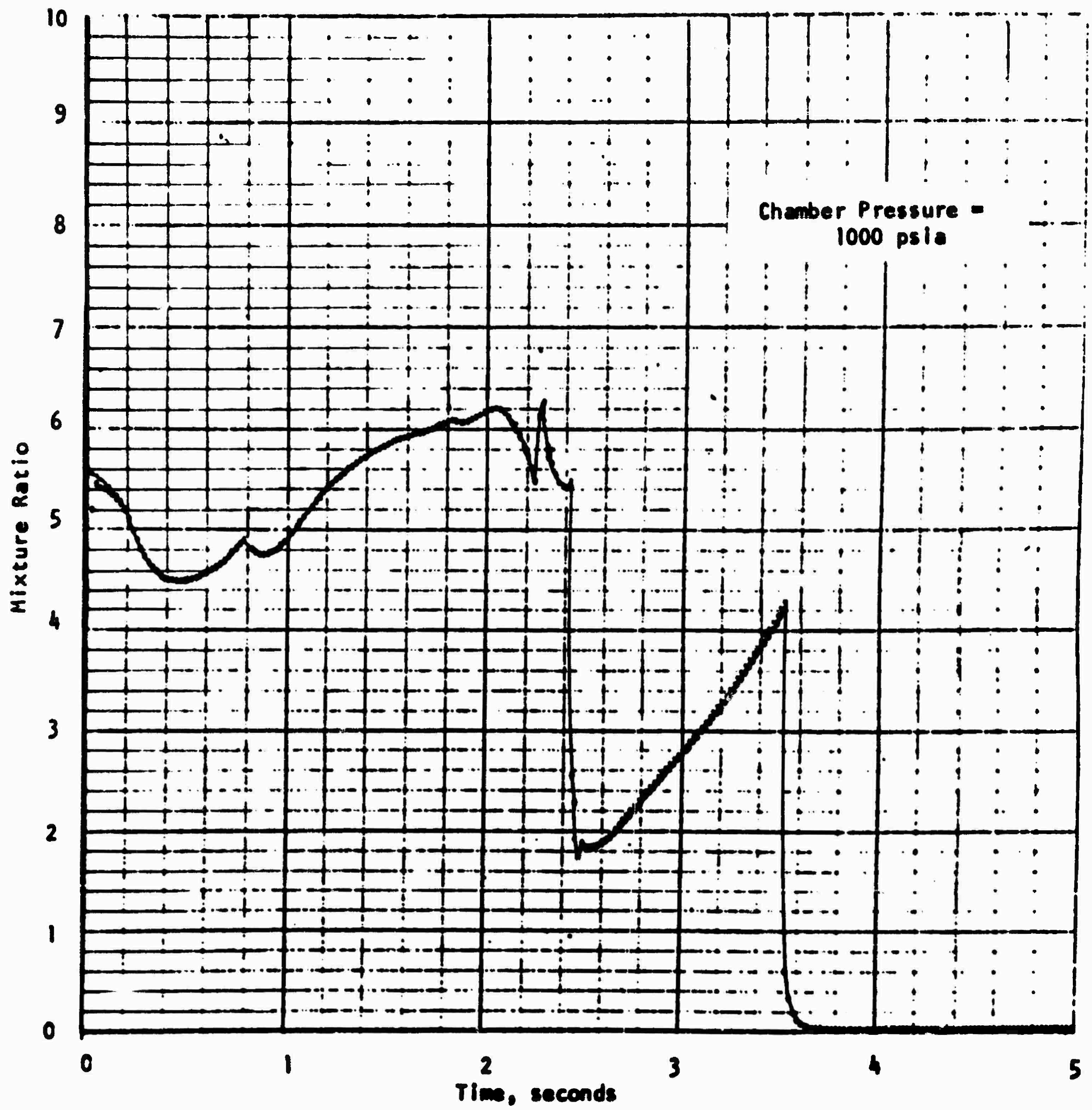


Figure 34. Mixture Ratio Versus Time for Cutoff Simulation

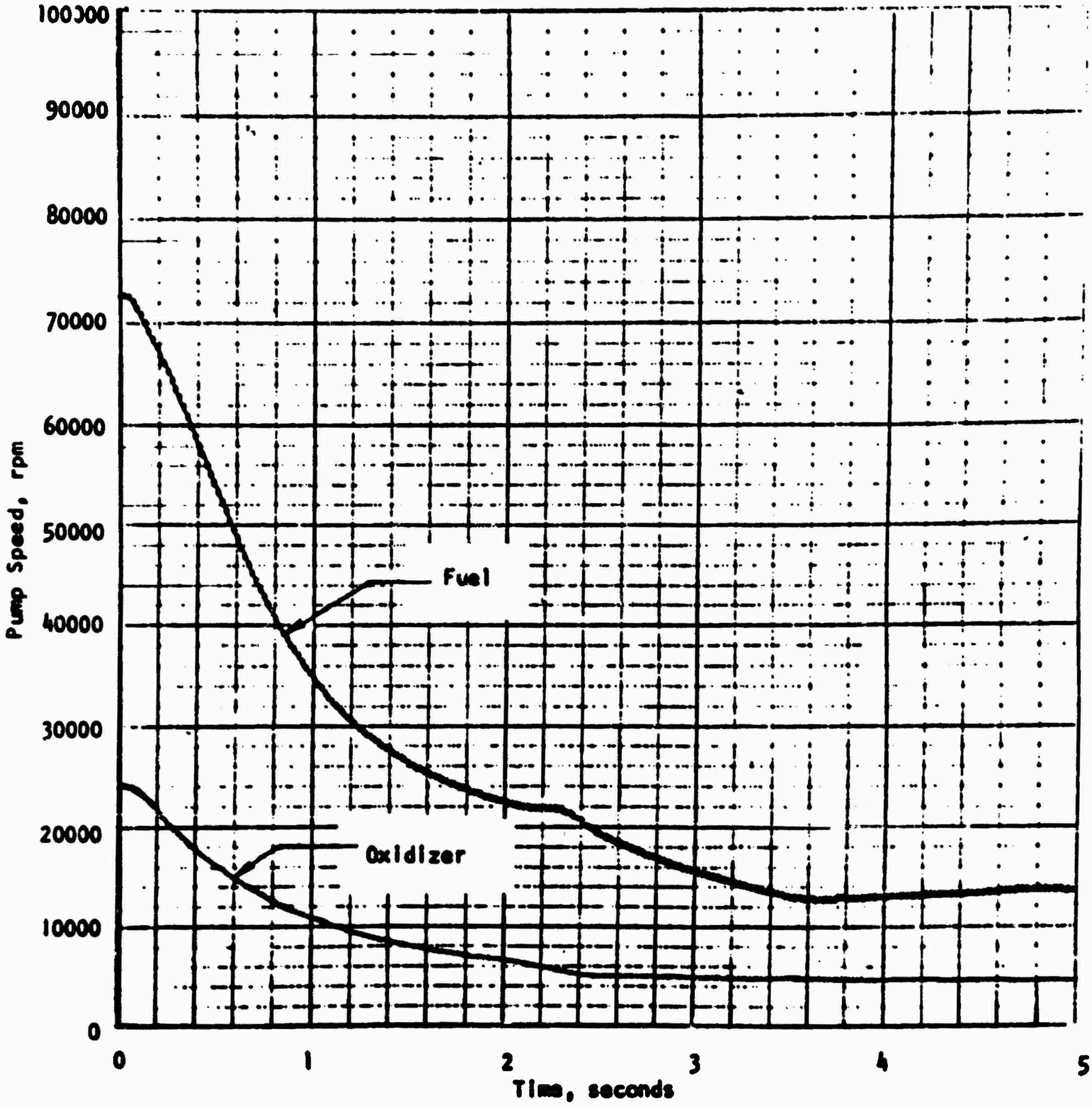


Figure 35. Pump Speeds Versus Time for Cutoff Simulation

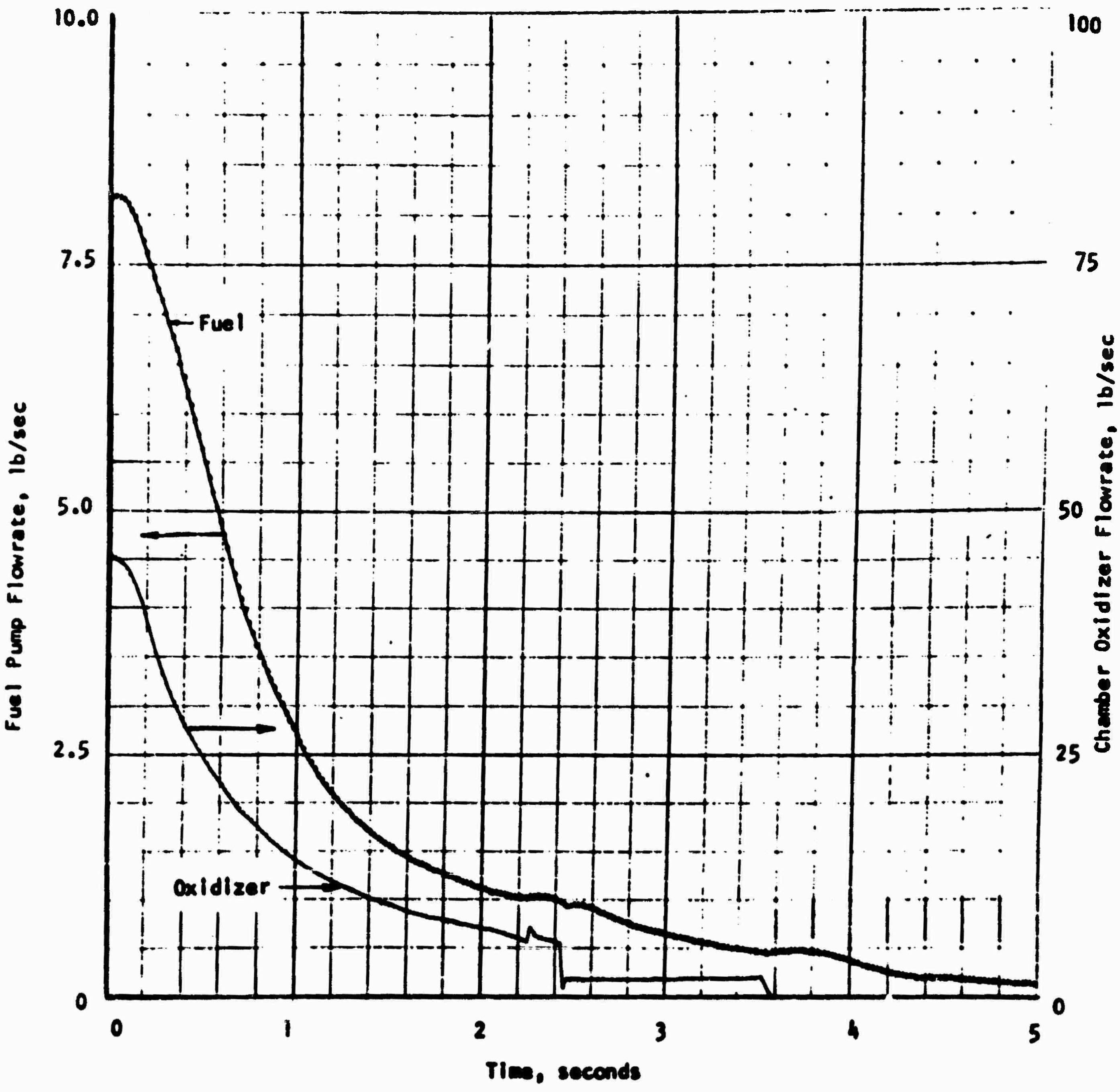


Figure 36. Propellant Flowrates Versus Time for Cutoff Simulation

Single-Panel Engine Transient Operation

The single-panel engine start and cutoff simulation was based on the same nominal conditions and control approach as the double-panel engine. An initial hardware temperature of 400 R was used.

Engine Start. Engine control sequencing for start is described in Fig. 37. The two control valves (turbine bypass and oxygen turbine inlet) have proportional plus integral control:

$$\text{Position} = K_{1*} \text{ Error} + K_2 \int \text{Error } dt$$

No cross compensation between the two valves was used.

The early portion of the start is accomplished open loop. After 3.5 seconds of operation, engine operation is turned over to closed-loop control. Thrust buildup takes about 6 seconds (Fig. 38), slightly longer than for the double-panel engine because of the greater fuel pump inertia.

In the start sequence (Fig. 37), the main fuel valve opens at engine start signal. With the oxidizer turbine inlet and the turbine bypass valves closed, initial power to the fuel turbine is provided while allowing the hydrogen flow to begin cooling the fuel pump (Fig. 39), high-pressure duct, and thrust chamber. The heat transfer in the fuel pump reduces the density of the hydrogen in the pump, thereby reducing fuel pump outlet pressure at any given fuel pump speed.

The main oxidizer valve is opened between 1.9 and 2.0 seconds after start signal. Between 1.9 and about 4.0 seconds, the oxidizer flow is priming the volume between the inlet valve and the thrust chamber and chilling the oxidizer pump. By 4.0 seconds, adequate fuel pump speed (Fig. 40) has been obtained to prevent high chamber mixture ratio when the oxidizer side is primed. Figure 41 shows a thrust chamber mixture ratio of about 1.0 at 4.0 seconds. Two mixture ratios are shown on Fig. 41. The engine mixture ratio is based on pump flowrates and

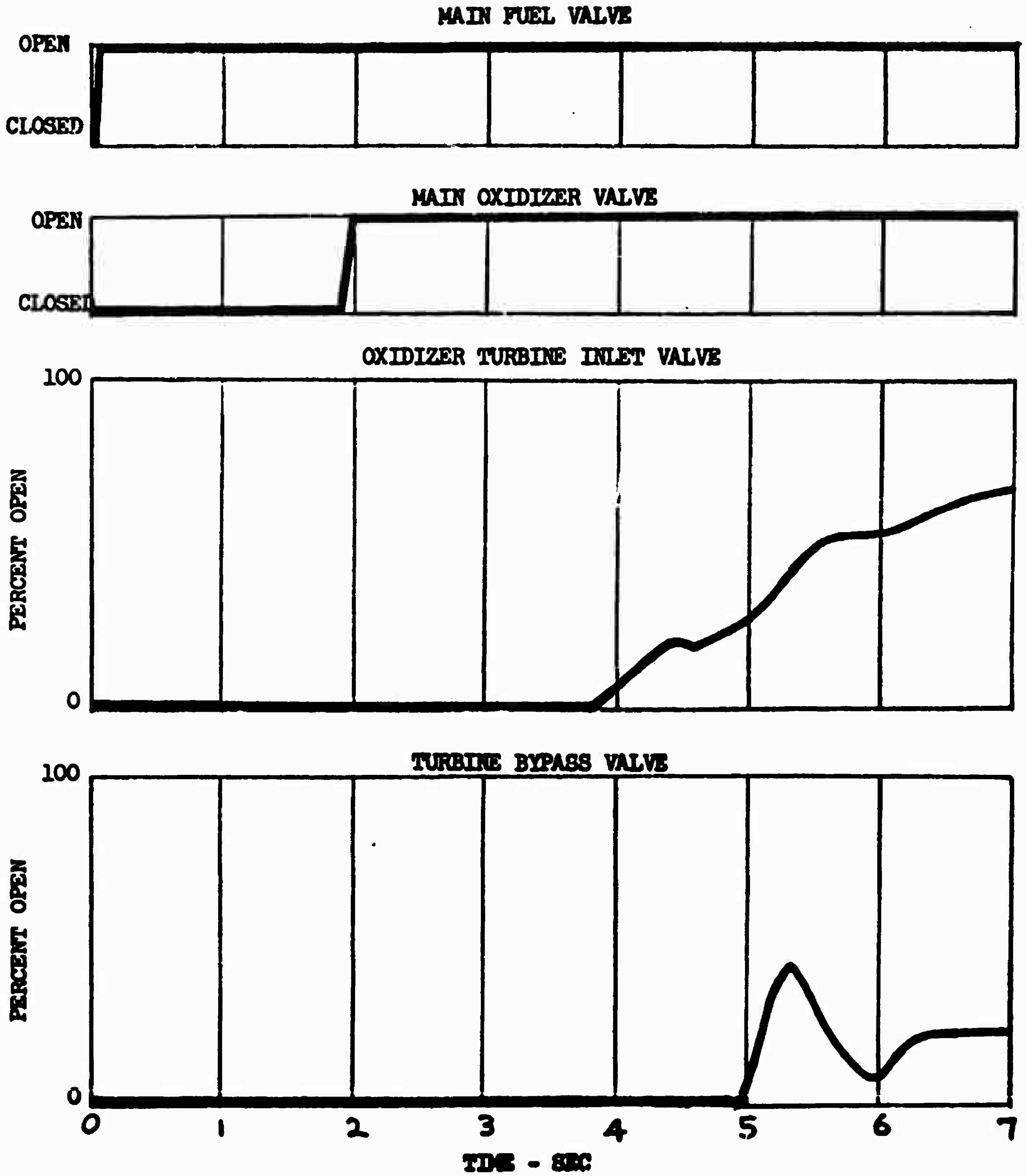


Figure 37. Engine Start Sequence

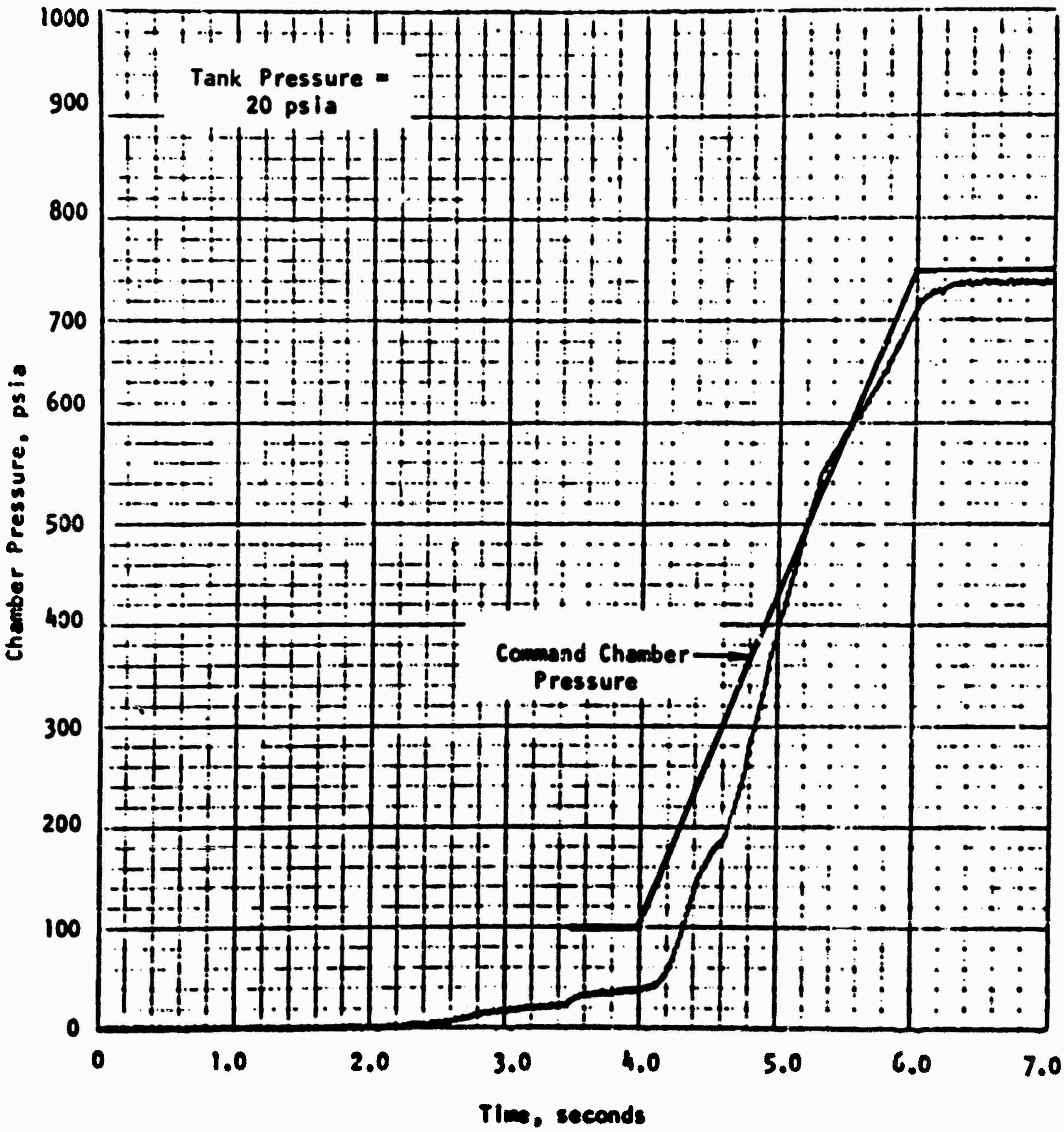


Figure 38. Thrust Chamber Pressure vs Time for Start Simulation

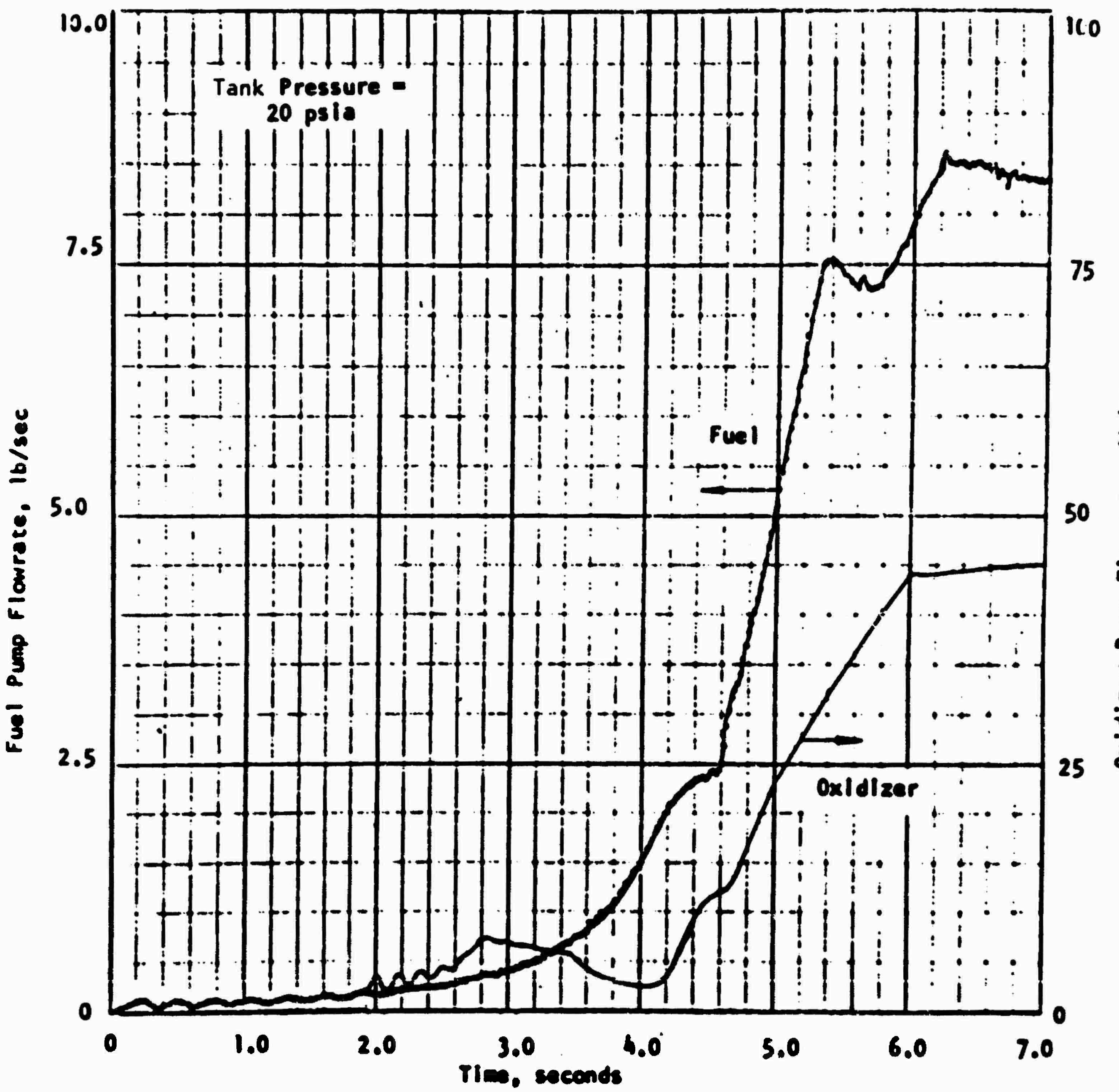


Figure 39. Pump Flowrate Versus Time for Start Simulation

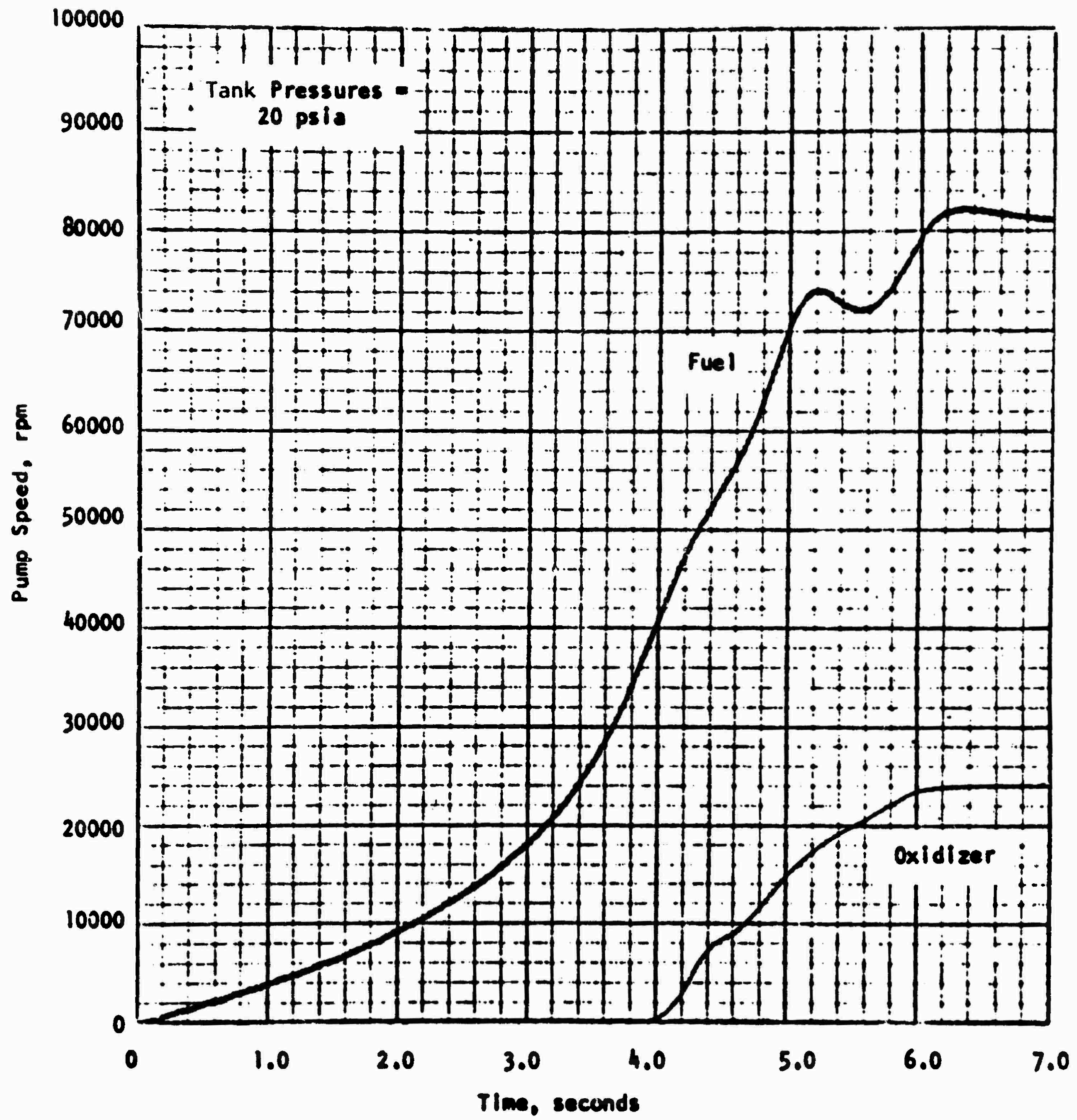


Figure 40. Pump Speeds Versus Time for Start Simulation

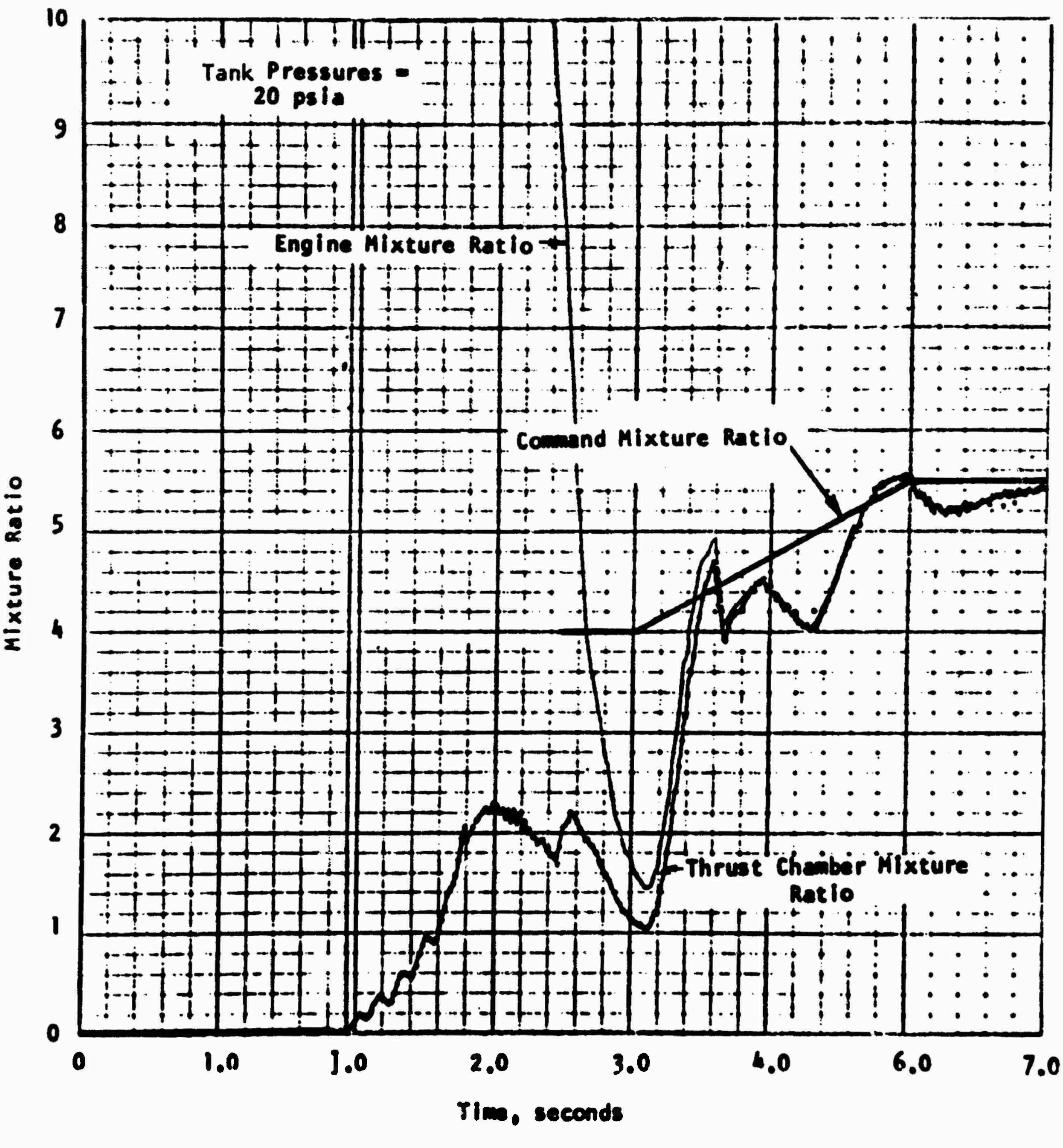


Figure 41. Mixture Ratio Versus Time for Start Simulation

the thrust chamber mixture ratio is based on injector flowrates. The primary difference is caused by the oxidizer priming. Initially, the oxidizer pump flow is high with only small amounts of oxidizer injector flow. Therefore, between 1.9 and 4.0 seconds, the thrust chamber is operating at low mixture ratios. At 3.5 seconds, the engine operation is turned over to closed-loop control.

Command chamber pressure (Fig. 38) is 100 psia until 4.0 seconds, and is then ramped to 750 psia in 2.0 seconds. The turbine bypass valve stays closed until the actual chamber pressure nears the command pressure. At this time (4.95 seconds) seconds), the lead compensation on the bypass valve causes it to begin opening. It then controls chamber pressure to the mainstage operating level.

The mixture ratio command (Fig. 41) is 4.0 until 4.0 seconds, and then is ramped to 5.5 at 6.0 seconds. When the engine mixture ratio drops below 4.0 at 3.7 seconds, the oxidizer turbine valve begins to open, causing the oxidizer turbopump to begin accelerating.

Engine Cutoff. Engine cutoff is accomplished, using the open-loop sequence shown in Fig. 42 . The turbine bypass valve is ramped to full open in 0.3 second and the oxidizer turbine inlet valve is closed in 0.4 second. This causes both pump speeds to decay. At 0.4 second after cutoff signal, the main oxidizer valve begins to close and reaches full closed in 0.3 second. The oxidizer purge is turned on at 0.5 second, and generates 60 psia at the oxidizer pump inlet. At these low oxidizer flowrates, there is very little pressure drop in the system so that chamber pressure is also about 60 psia (Fig. 43). By 0.8 second after cutoff signal, the oxidizer is purged from the pump.

The recommended purge is 60 psia at the oxidizer pump inlet. This purge level should be as high as possible to minimize the cutoff time. The 60-psia purge produces an oxidizer injector flow of about 2.0 lb/sec (Fig. 44), requiring 3 to 4 seconds to purge the oxidizer system completely.

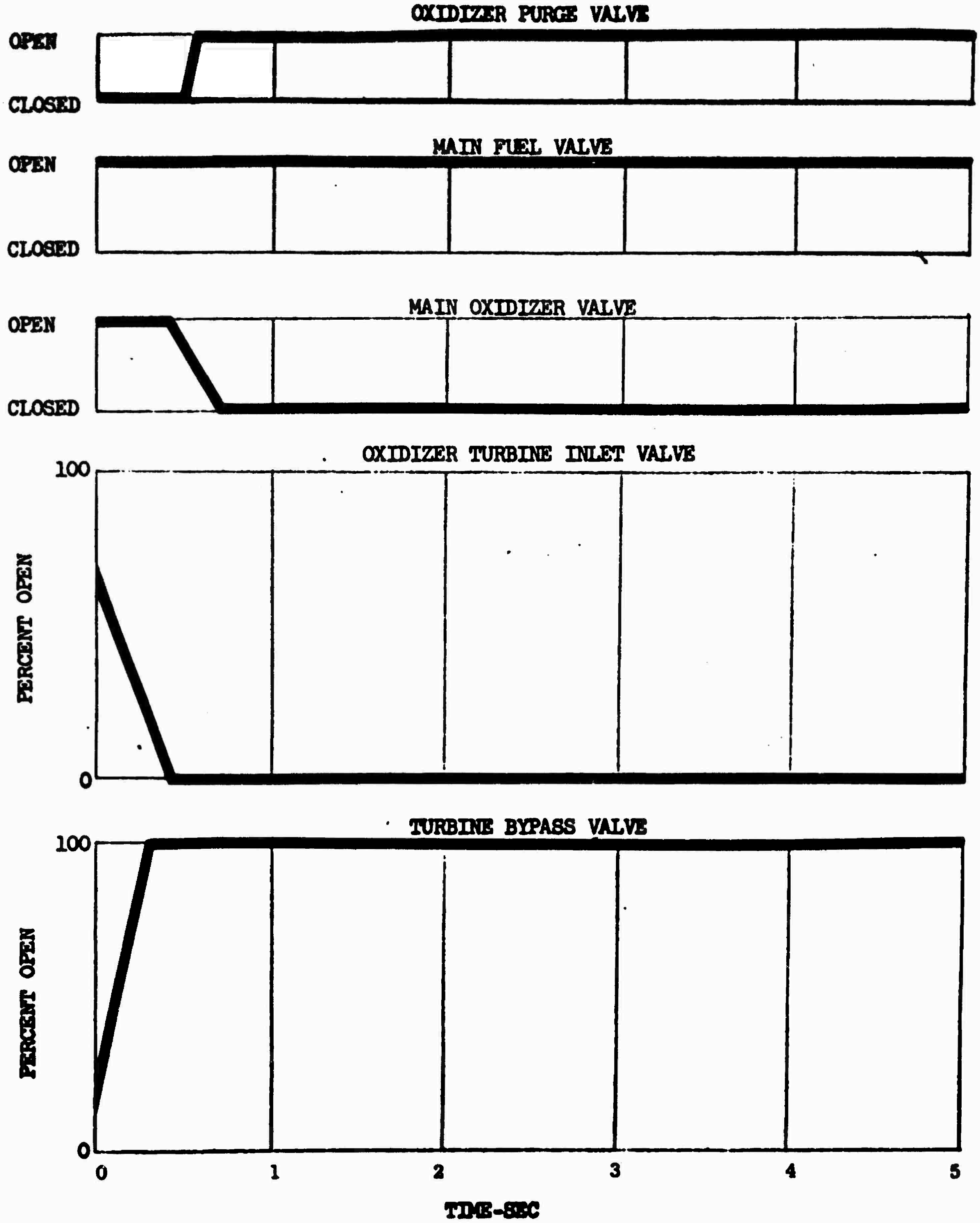


Figure 42. Engine Cutoff Sequence

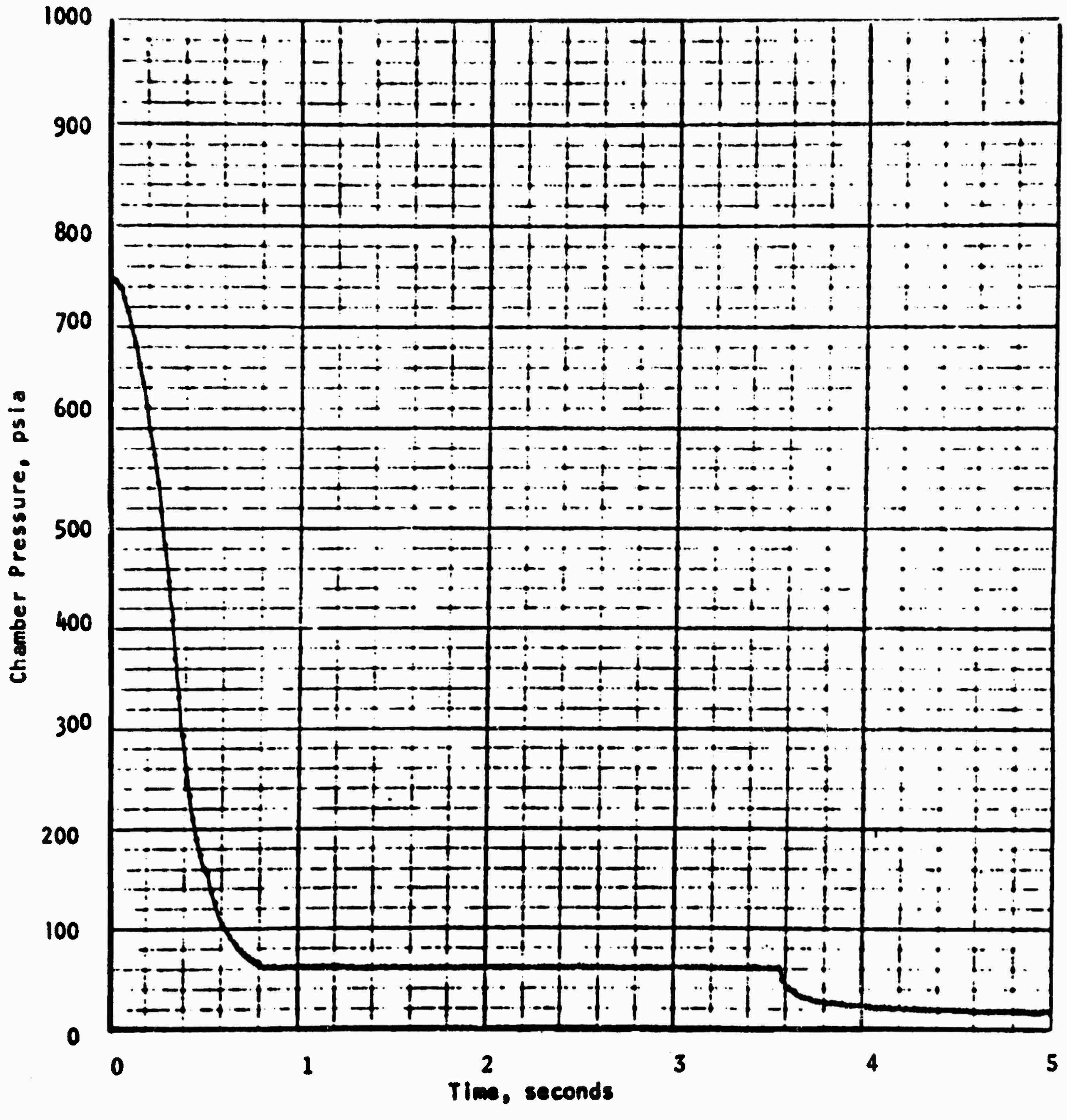


Figure 43. Thrust Chamber Pressure Versus Time for Cutoff Simulation

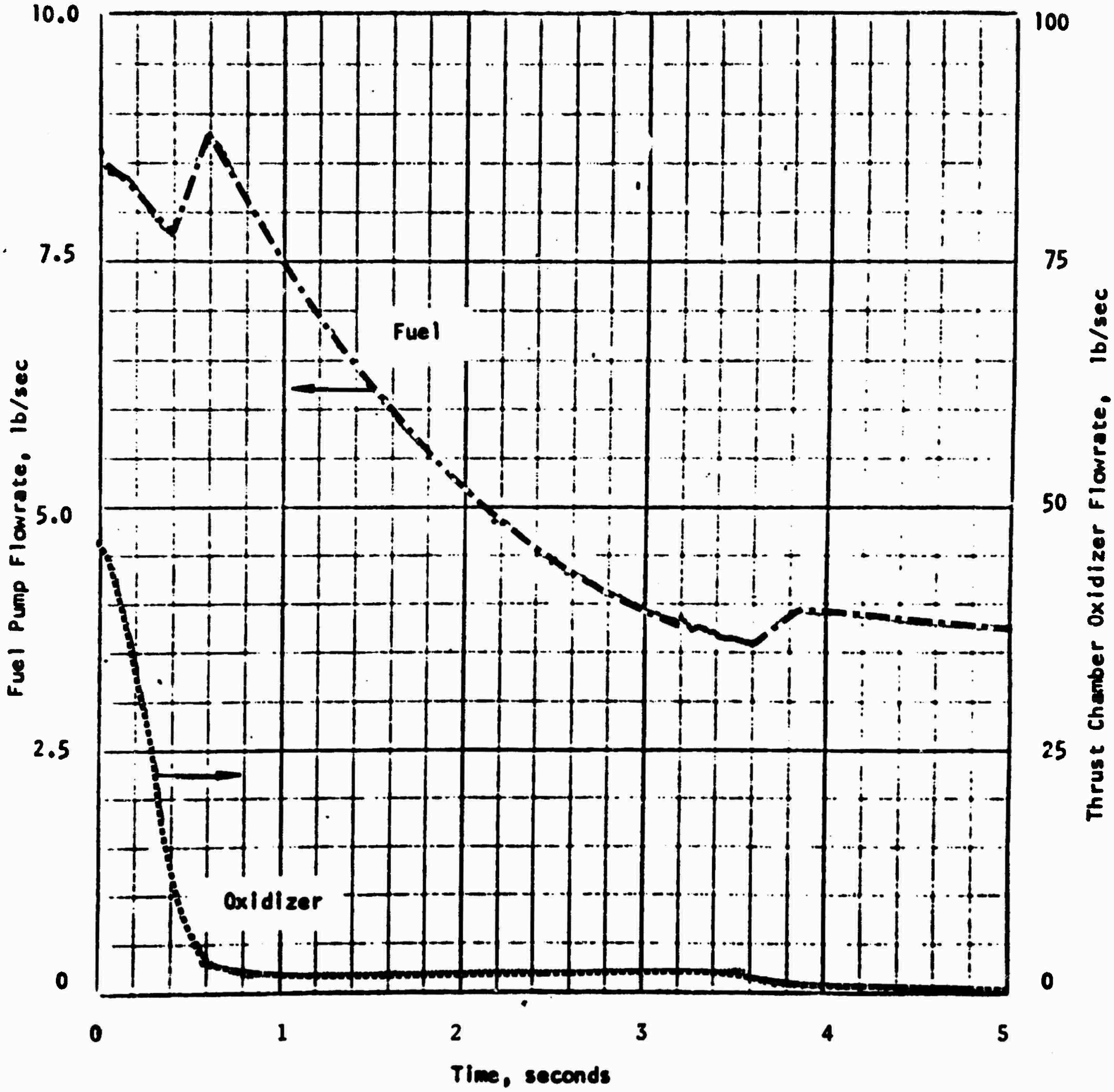


Figure 44. Propellant Flowrates Versus Time for Cutoff Simulation

The main fuel valve is closed as soon as the oxidizer system is completely purged. The corresponding pump speeds and thrust chamber mixture ratio traces are shown in Fig. 45 and 46.

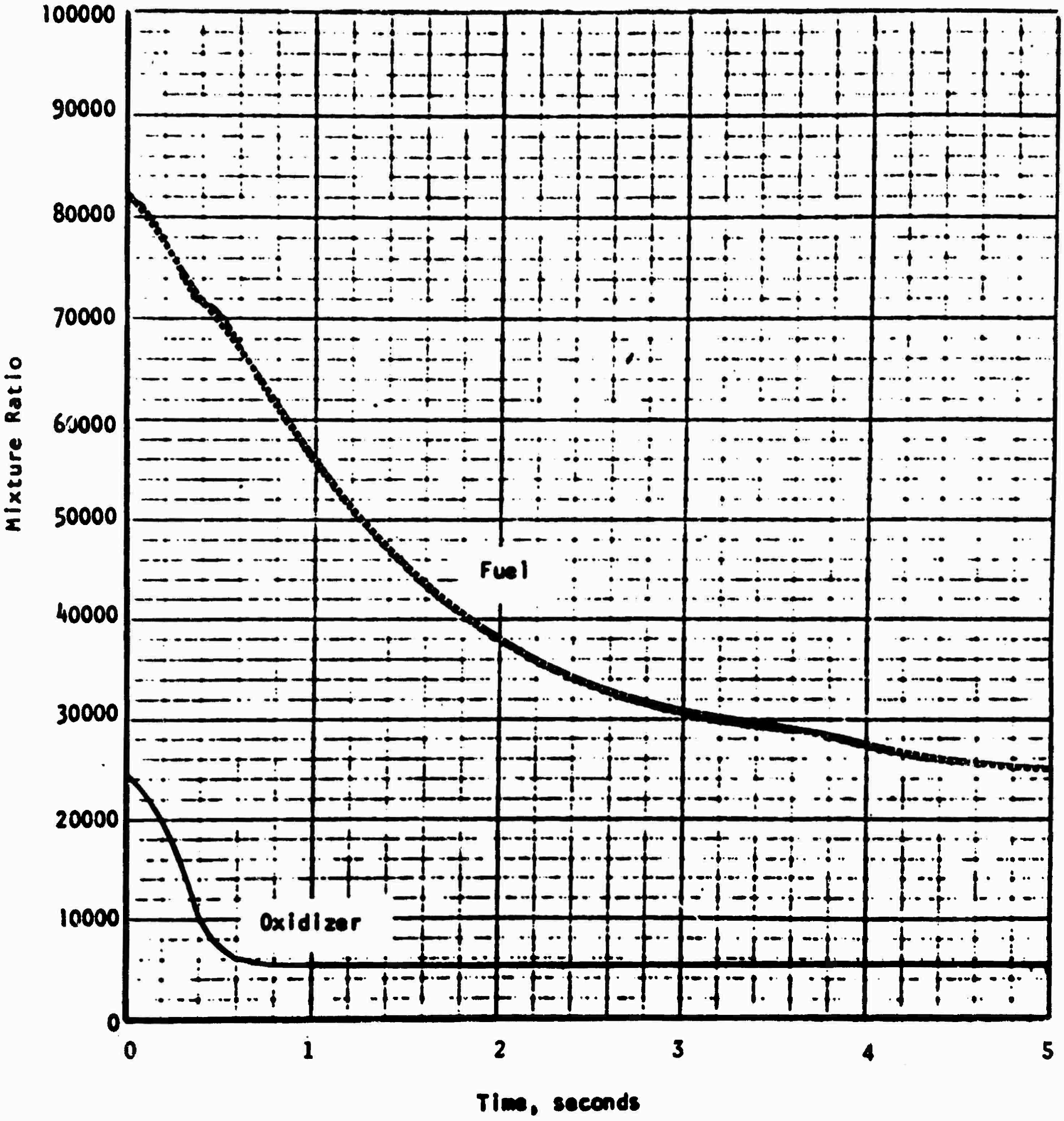


Figure 45. Pump Speeds vs Time for Cutoff Simulation

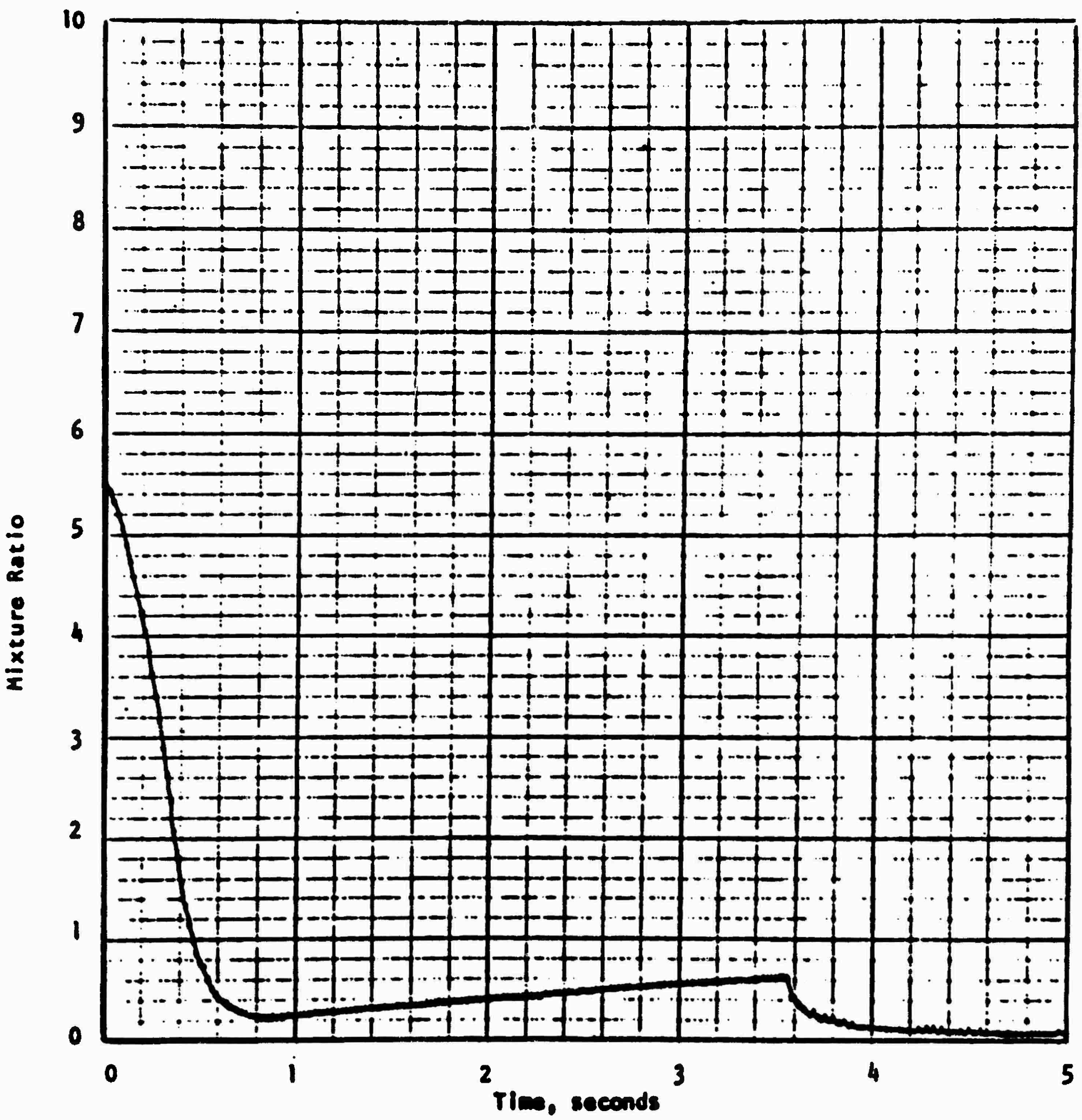


Figure 46. Thrust Chamber Mixture Ratio vs Time for Cutoff Simulation

FAIL-SAFE OPERATIONAL CAPABILITY

The capability for fail-safe operation of the O_2/H_2 AMPT aerospike engine was evaluated and the results are presented herein. The purpose of the analysis is to determine if the design can be fail-safe, and present operational requirements or engine system changes which may be necessary to ensure fail-safe operational capability.

For the purpose of this study, the engine is considered to be fail-safe when the failure of an engine component will not result in damage to anything but the engine. Therefore, the stage tank assembly and stage payload will not be in danger.

The failure mode and effect analysis (FMEA), presented in Appendix B identifies the potentially dangerous failures (those that may damage tank assembly or payload). These dangerous failures are listed in Table 29. For each failure, the actions required to ensure fail-safe operation were determined and also are listed in Table 29. This evaluation shows the engine can have fail-safe operational capability.

FAIL-OPERATIONAL CAPABILITY

A fail-operational capability was defined as the capability of providing a safe return of the vehicle to base despite a mission abort caused by a single failure. In the worst case, the engine may be required to operate at its nominal performance level and maintain the capability to operate throughout its nominal operating envelope. The results of the FMEA are studied to determine what type failures could occur and the changes that would be required to the engine to provide a fail-operational capability.

TABLE 29. FAIL-SAFE OPERATI

COMPONENT	POSSIBLE DANGEROUS FAI & FAILURE EFFECT
ENGINE SUBSYSTEM	
Gimbal Bearing & Thrust Mount Propellant Discharge Ducts & Manifolds Turbine Supply Ducts and Manifolds Turbine Discharge Ducts & Manifolds Heat Exchanger	Structural failure (not consi
CONTROL SYSTEM	
Main Oxygen Valve	Fails to close at cutoff caus cutoff. Possible Fire. Excessive through leakage cau hazard or hard start because oxidizer.
Main Hydrogen Valve	Fails to open at start causin oxidizer-rich start. Possibl
Bypass Valve Throttle Valve	Probably no dangerous failure
Pneumatic Pressure Regulator	No dangerous failures.
Low-Pressure Relief Valve	No dangerous failures.
Pneumatic System Isolation Valve	No dangerous failures.
Oxidizer System Purge Solenoid Valve	Fails to open during start al enter oxidizer system & combu enters.
Fuel System Purge Solenoid Valve	No dangerous failures.
Purge Check Valves	Failure of oxidizer system pu to open. No oxidizer purge d allowing fuel to enter oxidiz bust when oxidizer enters.
Pressurant Check Valves	No dangerous failures.

SAFE OPERATION RECOMMENDATIONS

NGEROUS FAILURE E EFFECT	PREVENTION OF DANGEROUS FAILURE
(not considered in FMEA)	A structural failure from improper fabrication or assembly of the engine will be discovered in the engine system checkout test & the problem corrected before engine delivery. These components are considered to be fail-safe.
cutoff causing LOX-rich Fire.	To assure fail-safe operation a stage prevalve, programmed closed immediately following cutoff, should be close to the engine to limit the oxidizer supply to a possible fire.
leakage causing fire art because of accumulated	If necessary, the oxygen system [illegible] can begin before the normal start sequence signal in [illegible] to purge accumulated oxidizer from the system.
start causing attempted rt. Possible fire.	If engine controller has to receive nominal start hydrogen valve position signal &/or hydrogen system pressure signal before oxidizer valve is opened, no oxidizer-rich start will occur.
rous failures.	Failures of these turbine flow control hot hydrogen valves will result in irregular engine operation. Engine sensors should detect these irregularities & signal engine cutoff before a dangerous failure occurs.
ures.	
ures.	
ures.	
ing start allowing fuel to stem & combust when oxidizer	If nominal purge pressure signal is required before oxidizer valve can be signaled open, this failure will not occur.
lures.	
zer system purge check valve lizer purge during start enter oxidizer system & com- r enters.	If nominal purge pressure signal is required before oxidizer valve can be signaled open, this failure will not occur.
lures.	Failure to open. Loss of tank pressure will signal engine cutoff before dangerous failure could occur.

2

TABLE 29. (Cor

COMPONENT	POSSIBLE DANGEROUS FA & FAILURE EFFECT
TURBOPUMP ASSEMBLIES	
Oxygen Turbopump Assembly	Fails during operation from due to rubbing of inducer or
	Excessive shaft seal leakage explosion hazard in seal cav gen & oxygen mixing.
Hydrogen Turbopump Assembly	No dangerous failures.
THRUST CHAMBER ASSEMBLY	
Injectors Combustion Chambers Nozzle & Base Closure	Probably no dangerous failur
Combustion Wave Igniter System	One or more elements fail to fire or explosion from unlit lants.

29. (Concluded)

NGEROUS FAILURE URE EFFECT	PREVENTION OF DANGEROUS FAILURE
tion from oxidizer explosion inducer or impeller.	This type of failure from improper fabrication or assembly of the engine will be discovered during the engine system checkout test and the problem corrected before engine delivery.
al leakage causing fire or n seal cavity due to hydro- g.	More than one seal in the seal package must fail before separation of the hydrogen & oxygen is inadequate. Pressure &/or temperature measurements in drain cavity or drain line can be used to monitor seal leakages & cut off the engine when excessive leakage is indicated.
res.	
ous failures.	Injector obstruction, chamber leakage, or nozzle leakage should be noticed in performance degradation & the engine shut down before a dangerous failure occurs.
ts fail to ignite. Possible from unlit segment propel-	Experience during engine development program may show that propellants from unlit segment will burn harmlessly until engine cutoff can be signaled or that the chance of only part of the segments igniting is remote. If not, redundant ignition systems will be required to make engine fail-safe.

2

The thrust mount, ducts, and manifolds are considered to have only structural failures. These failures, and the structural failure mode of the other engine components and subsystems can be effectively eliminated by checkout-testing each engine at the engine operating point, which results in the highest loads the engine would encounter during mission operation.

For complete assurance of fail-operational capability, the main inlet valves and the hot hydrogen turbine control valves could be replaced by series-parallel valve sets. This replacement of each individual valve by a four-valve set will noticeably increase engine weight and cost. These valves could, as an alternative, be given some fail-operational characteristics by providing a redundant actuation method to prevent operational failure due to this component function. The valve position instrumentation also could be redundant to ensure that the valve actuation characteristics monitored during engine readiness checkout have accurately monitored the valve condition.

The pneumatic system could be made fail-operational by replacing each of the individual components by series-parallel component sets. Though this requires a large number of components, it allows the system to be fail-operational without requiring redundant helium supply tanks.

Providing redundancy for components as major as the turbopump assemblies does not seem practical for an engine system of this type. The turbopump assemblies will, therefore, not have complete fail-operational capability. The shaft seal packages in the nominal turbopump designs already provide adequate separation even with a single seal failure. The use of overstress checkout testing and extensive instrumentation to monitor turbomachinery condition would have to be relied upon to eliminate other types of failures.

Once structural integrity and satisfactory performance have been demonstrated during engine checkout testing, the thrust chamber assembly should be highly reliable if the remainder of the engine operates satisfactorily. However, if a significant failure does occur, nothing practical can be done to ensure continued normal operation of the engine system. To minimize the possibility of failure, cverstress checkout testing and reduced thermal cycles between inspections and overhauls could be utilized.

To provide ignition system fail-operational capability, the ignition system controls could be replaced by series-parallel component sets. The nominal design already provides redundant spark ignition for the premix chamber.

The engine avionics system could incorporate redundant elements to provide a fail-operational/fail-safe capability. No electronic single-point failure would result in loss of avionic function, engine shutdown, or an unsafe operating condition.

ENGINE MAINTENANCE

Maintenance Analysis

The maintenance plan for the engine was established based on the engine life cycle illustrated in Fig. 47 . The engine is service-free and has no routine maintenance after an operational mission. Scheduled maintenance and refurbishment occurs after 60 vacuum starts or 2 hours of in-flight operation. This activity costs less than 5 percent of a new engine cost. A scheduled overhaul activity occurs after 300 vacuum starts or 10 hours in-flight operation with the cost not exceeding 25 percent of a new engine cost. The total engine service life is 50 hours or 1500 vacuum starts.

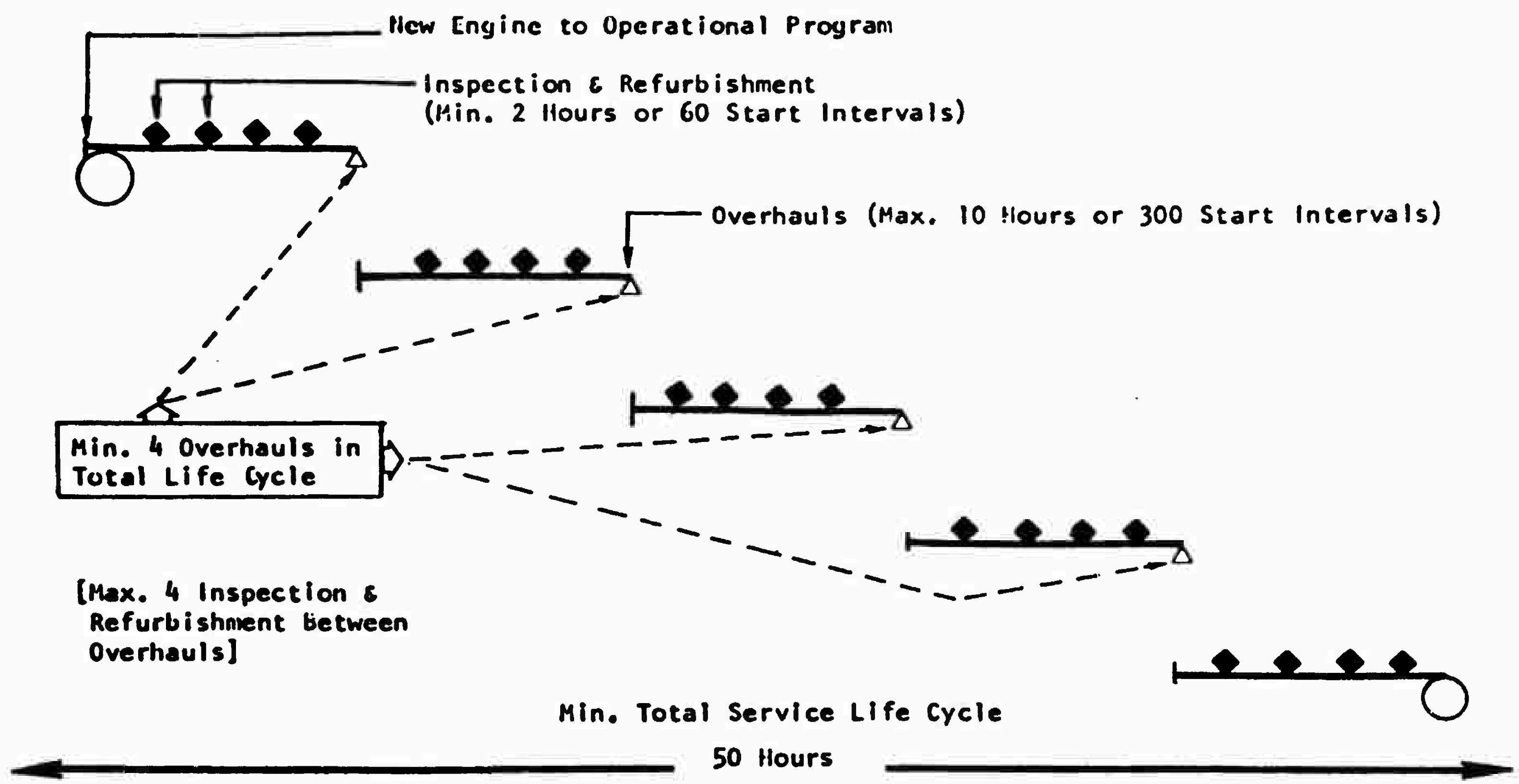


Figure 47. Engine Life Cycle Time-Life

Maintenance of the engine is based on:

1. Component design for the total service life unless a serious engine performance compromise would be incurred.
2. Engine design features resulting in high operational reliability; internal inspection of critical areas without disassembly and simplified component replacement of Line Replaceable Units (LRU)
3. Continuous data assimilation and review (for trend analysis and maintenance action determination) from automated preflight checkouts, in-flight engine condition monitoring and fault isolation, and postflight inspections and checkout
4. Utilization of ground support equipment whose suitability has been well verified by use as an integral part of the engine development and manufacturing processes
5. An overall maintenance plan designed to ensure safe vehicle operation with minimum turnaround effort between flights.

Engine Design for Low Maintenance Cost

Major engine component design life expectancy is summarized in Table 30. With the exception of the pump seals and bearings, the thrust chamber segment, and the spark igniters in the combustion wave generator, all components are expected to meet the total engine service life. These remaining components need no scheduled maintenance or overhaul. Pump seals and bearings and the thrust chamber segments are designed for 10 hours and 300 cycle life to avoid significant reduction in engine performance (see Design Condition Variation Section). The spark igniters in the combustion wave generator will be changed at each overhaul because the cost is minor and ignition assurance is increased.

The engine assembly utilizes welded joints to eliminate leakage and delete many of the leak-detection requirements that were time-consuming on the

TABLE 30. ENGINE SYSTEM MAJOR
COMPONENT LIFE EXPECTANCY SUMMARY

Component	Minimum Life Between Overhauls	
	Starts	Hours
Thrust Chamber Assembly		
Segment (liner and injector)	300	10
Structural Rings	Indefinite	Indefinite
Tie Bolts	1500	50
Primary Nozzle	1500	50
Base Closure	1500	50
Thrust Mount and Bolts	Indefinite	Indefinite
Oxidizer Turbopump Assembly		
Pump Shaft	1500	50
Pump Impeller	1500	50
Bearings	300	10
Seals	300	10
Housing	Indefinite	Indefinite
Turbine Wheel/Blades	1500	50
Fuel Turbopump Assembly		
Pump Shaft	1500	50
Pump Impeller	1500	50
Bearings	300	10
Seals	300	10
Housing	1500	50
Turbine Wheel/Blades	1500	50
Flow Control Valves		
Main Fuel Valve	1500	50
Main Oxidizer Valve	1500	50
Oxidizer Turbine Inlet Valve	1500	50
Oxidizer Turbine Bypass Valve	1500	50
Miscellaneous Pneumatic Valves	1500	50
Spark Igniter Assembly		
In Premix Chamber	300	10
Gimbal Bearing Assembly	1500	10
Avionics Controller	Indefinite	Indefinite
Manifolds & Ducts	Indefinite	Indefinite
Heat Exchanger	Indefinite	Indefinite

Saturn/Apollo systems. During the engine development program, the leak-free operability of the engine, and thrust chamber assembly in particular, will be demonstrated. Critical weldments in the power package area and thrust chamber assembly will be monitored for structural integrity during the scheduled refurbishment cycle.

External, visual detection of leakage is made routine in inspection procedures through the use of color change tapes and/or elastomeric material applied to each engine joint. These simple detectors represent a negligible weight and cost per mission and are not subject to costly maintenance action common with electronic devices. An H_2 leak-detection tape was successfully tested on J-2 engines in 1967 at Rocketdyne (NAS8-19). Hydraulic leakage tracer tapes are available from aircraft usage. Elastomeric paint, potentially useful in all systems, is specifically applicable to O_2 and N_2 systems; this paint was conceived during the J-2 program at Rocketdyne (NAS8-19). Therefore, all welded joints can be inspected externally.

Based on recent advances in airline propulsion maintenance, provisions are made for internal inspection of components without removal from the engine. Borescope access ports and guides are provided on all major components. Rocketdyne has used borescopes in several previous programs. During the F-1 program (NAS8-18734), borescopes and "retrievescopes" (combined optics and articulated grasping mechanism) were used for oxidizer dome and heat exchanger inspection. The Atlas Sustainer program (AF04(645)-1 and follow-ons) used borescopes to inspect turbopump bearings suspected of overheating and the Atlas Booster program (AF04(645)-1 and follow-ons) used borescopes to inspect turbopump gears and turbine blades.

Component removal, when required for overhaul, is facilitated by providing adequate room for standard line cutting and welding equipment.

Engine Flight Monitoring

Low-cost operation of the engine in terms of maintenance actions is ensured by the performance of automated on-board checkout to identify parts for replacement, the verification of engine conditions by monitoring performance in flight, and the acquisition of maintenance data for vehicle recording and ground processing for maintenance trend analysis and preventive maintenance determination. The avionics package (which can be either a part of the vehicle centralized avionics capability or an engine subassembly) provides the elements for sending and monitoring engine performance. The engine avionics sensors provide engine data for performance control, engine readiness checkout, limit control monitoring, and maintenance recording and condition checkout monitoring. Approximately 50 engine parameters, consisting of pressure, temperature, propellant volumetric flowrate, turbopump shaft speed, vibration and position, are measured (Table 31). The controller performs the computations (using digital computers within the assembly) for engine control and sequencing, checkout, and monitoring. The engine-related avionics system incorporates redundant elements to provide a fail-operational/fail-safe capability. No electronic single-point failure will result in loss of avionic function, engine shutdown, or an unsafe operating condition.

Ground-Based Maintenance

The sequence of operations associated with the ground-based engine maintenance plan is summarized in the block diagrams shown in Fig. 48 and can be viewed as the following key paths:

1. Normal Turnaround, which includes the launch preparations, mission, and postflight operations in a "go" condition, i.e., no scheduled maintenance activity is required.
2. Scheduled Inspection, which includes the Inspection and Refurbishment activity loop at the engine service area within the flight operations base, whether normally scheduled or because of a "no-go" mission condition.

TABLE 31. ENGINE RECORDED PARAMETER MEASUREMENT LIST

	Function*					
	1	2	3	4	5	6
FUEL TURBOPUMP						
Pump Inlet Pressure	x			x		
Pump Discharge Pressure	x			x		
Pump Inlet Temperature	x			x		
Pump Discharge Temperature	x			x		
Fuel Flowrate	x	x	x	x		x
Shaft Speed				x		x
Turbine Inlet Pressure				x		
Turbine Inlet Temperature				x		
Acceleration				x		x
OXIDIZER TURBOPUMP						
Pump Inlet Pressure	x			x		
Pump Discharge Pressure	x			x		
Pump Inlet Temperature	x			x		
Pump Discharge Temperature	x			x		
Oxidizer Flowrate	x	x	x	x		x
Shaft Speed				x		x
Turbine Inlet Pressure				x		
Turbine Inlet Temperature				x		
Acceleration				x		x
PNEUMATIC/HYDRAULIC SYSTEM						
Hydraulic System Pressure				x	x	
Helium Supply Pressure				x	x	
Helium Supply Temperature				x	x	
Fuel Duct Purge Pressure					x	
Oxidizer Duct Purge Pressure					x	
Oxidizer Pump Seal Purge Pressure					x	
Fuel Pump Seal Purge Pressure					x	

*Functions:

1 Engine Start & Cutoff Control
2 Engine Thrust Control
3 Mixture Ratio Control
4 Vehicle Performance Evaluation
5 Engine Ready
6 Engine Limit Control

TABLE 31. (Concluded)

	Function*					
	1	2	3	4	5	6
THRUST CHAMBER						
Combustion Chamber Pressure	x	x	x	x		x
Coolant Chamber-Circuit Exit Pressure			x	x		x
Coolant Chamber-Circuit Exit Temperature			x	x		x
Fuel Injection Pressure			x	x		x
Fuel Injection Temperature			x	x		x
Oxidizer Injection Pressure			x	x		x
Oxidizer Injection Temperature			x	x		x
Thrust Chamber Skin Temperature			x	x	x	x
Acceleration				x		x
FLOW CONTROL VALVE POSITIONS						
Main Fuel Valve	x			x	x	
Main Oxidizer Valve	x			x	x	
Oxidizer Turbine Inlet Valve (MR)	x		x	x	x	x
Oxidizer Turbine Bypass Valve (Throttle)	x	x		x	x	x
MISCELLANEOUS						
Base Bleed Pressure				x		x
Base Bleed Temperature				x		x
Avionics Controller Internal Pressure	x	x	x	x	x	x
Avionics Controller Internal Temperature	x	x	x	x	x	x
Ignition System Current	x			x	x	

*Functions:

1 Engine Start & Cutoff Control
2 Engine Thrust Control
3 Mixture Ratio Control
4 Vehicle Performance Evaluation
5 Engine Ready
6 Engine Limit Control

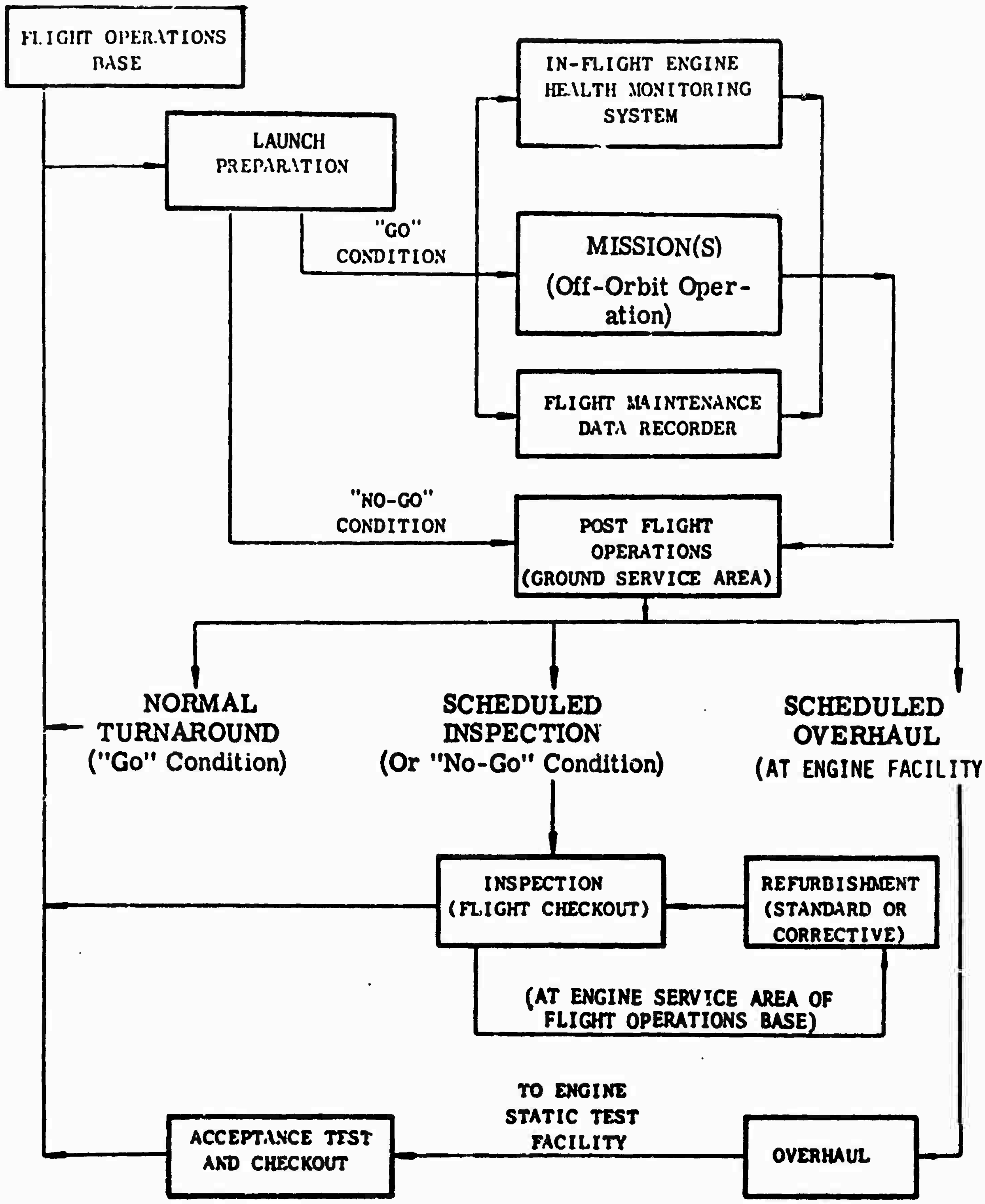


Figure 48. Ground-Based Engine Maintenance Plan (Sequence of Operations)

3. Scheduled Overhaul, which includes maintenance activity at the engine facility and subsequent acceptance testing at the engine static test facility prior to return to the flight operations base.

The general prelaunch and postflight checkout of the engine is automated and conducted as an integral part of the overall vehicle checkout procedure. Engine operating data are supplied to the flight maintenance data recorders. The flight monitoring capability can be provided by the vehicle-integrated avionics system. Data from the maintenance data recorders are used during each post-flight analysis to re-establish "go" conditions and to augment maintenance planning and scheduling of turnaround operations.

After return to the recovery area records are removed for analysis, and the safeing and drying operations are initiated. The engine safeing and drying is accomplished by purging the engine until humidity indicators show a dry condition. These "pre-maintenance" operations are accomplished in a special area designed for this type of activity. For a routine engine ground maintenance cycle (contrasted to a normal turnaround/no-maintenance cycle), Inspection and Refurbishment are the only planned operations.

<u>Scheduled Inspection</u>. A complete flight-type automatic checkout is the first step of the scheduled inspection activity. This checkout is conducted using the same vehicle systems (avionics package) that are used for automatic check out prestart in the space environment. These checkout results, analyzed by the on-board computer, are provided to the maintenance crew and to the trend analysis computer program in which all data from the engine are reviewed (automatic checkout and fault isolation), including data from earlier refurbishments and overhauls. This "refurbishment data review" emphasizes life-oriented data such as rotating machinery, combustion device accelerometers,

engine and component efficiency calculations, and basic line-joint and seal integrity. The analysis also makes use of a special file of wearout-type failures experienced on the engine during previous testing. This list is compared to data accumulated during the present refurbishment interval.

The Inspection procedure involves the engine as installed in the vehicle and occurs in the engine service area; external and internal inspection procedures are facilitated by full gimbal of the engine which provides access space from the top side of the engine (Fig.49). Because the base closure is fixed in design (nonremovable), all inspection activity occurs from the top side of the installation. Engine removal from the vehicle is necessary only when inspection procedures indicate the need for component removal.

The actions of the Inspection block (Fig. 48) are listed for major subsystems in Table 32 and associated costs are included in Tables 33 and 34 . The engine is inspected visually (externally) for signs of structural damage or deterioration, overheating or erosion, and for external wear due to contact of adjacent parts. Certain anomalies may exist which cannot be detected by the automatic checkout system and which require visual inspection for detection and evaluation (e.g., minor fluids leakage, structural failure by cracks or deformation local erosion in key combustion device components, externally caused damage, and foreign material or contamination).

Inspection of the thrust chamber assembly includes an internal segment liner and injector face check with access through the throat gap using a fiber optic borescope augmented with lights and mirrors. All key weldments are externally checked for leakage through the use of color change tapes or elastomeric paint; the backup structure bolts are torque-checked; the primary nozzle and base closure are visually inspected for deformation, bolt torque checking, and for dye-penetrant and zone radiography for suspect regions; and a leak check is made of the assembly.

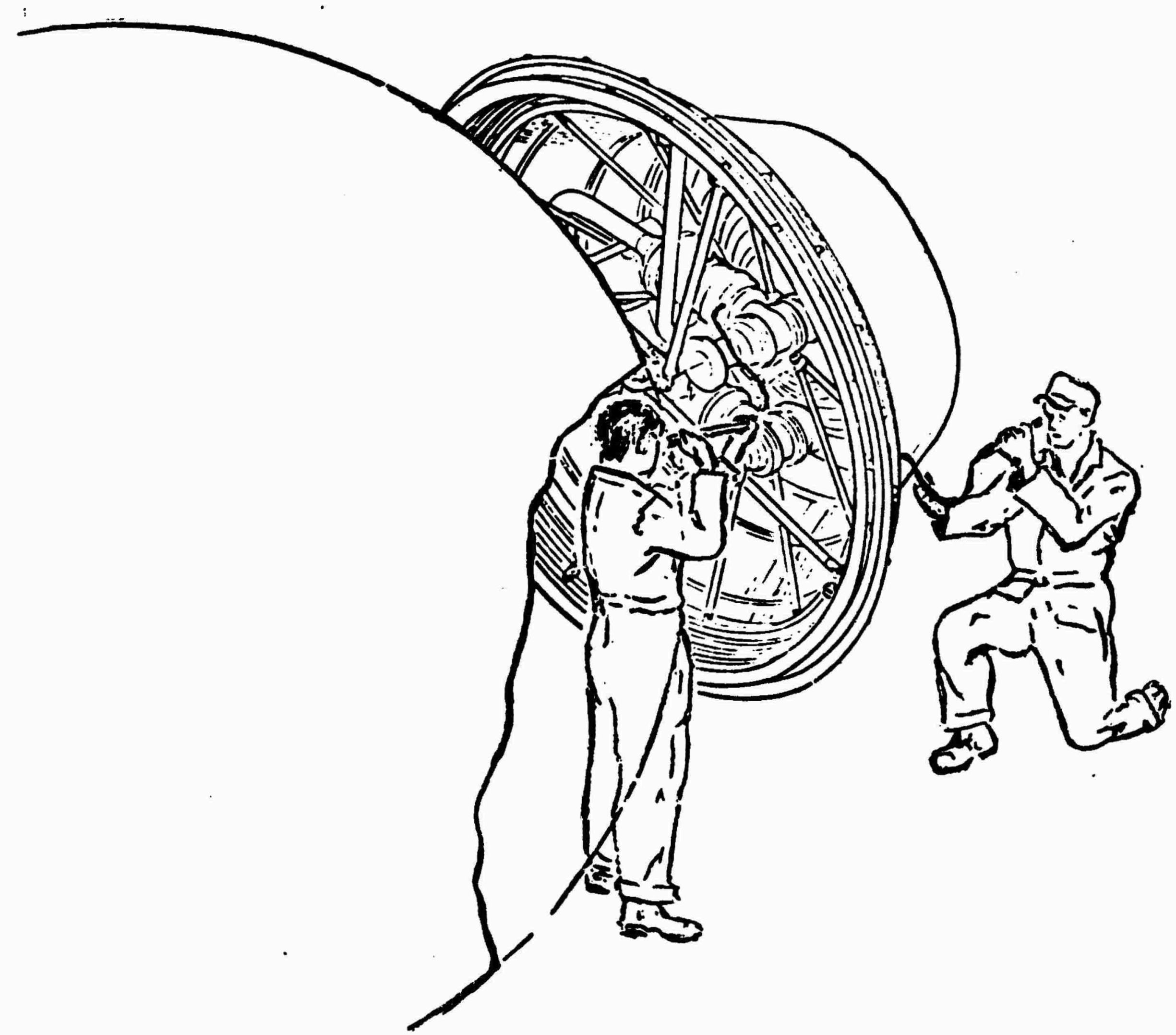


Figure 49. Ground Service Inspection

TABLE 32. ENGINE INSPECTION PROVISIONS (Major Subsystems)

Inspection Area	Type of Inspection	Type of Fault	Inspection Technique	Access
Thrust Chamber Assembly (TCA)	Internal-Segment Liner & Injector Face	Signs of Thermal Damage (erosion, cracking, plugging)	Flexible Fiber Optic Borescope Augmented w/ Mirrors & Lights	TCA-Throat Gap
	External-Weldments, Ducts, Manifolds & Lines	Leak Check	Lights & Color Change Tapes or Elastomeric Paints	TCA-as installed
	External-Bolt Torque Check	Structural Integrity	Torque Wrench on Back-up Ring Bolts	TCA-as installed
	Internal-Spark Ignition (automatic checkout)	Readiness	Provide Current to Igniters	TCA-automatic
	Automatic Checkout	Readiness	Comparison to Hot-Fire Data History	TCA-automatic
	External-Primary Nozzle & Base Closure	Tube Cracks, Splits, Holes	Visual, Mirrors & Lights	TCA-as installed
	External-Thrust Structure	Deformation & Structural Integrity	Bolt Torque Check, Tolerance Measurement, & Dye Penetrant w/Zone Radiography	TCA-as installed
	External-TCA	System Leak Check	Plug Throat & Pressurize Subsystems	TCA-as installed
Turbopump Assemblies (TPA)	Internal-Bearings	Signs of Thermal Damage, Spalling	Flexible Fiber Optic Borescope w/Special Guides;Use of Audio-Fault Detection Also Possible	TPA-speed. Speed pickup instrn. port

TABLE 32 (Concluded)

Inspection Area	Type of Inspection	Type of Fault	Inspection Technique	Access
Turbopump Assemblies (TPA) (cont'd)	Internal-Seals	Leak Check (functional) & Readiness (automatic)	Pressure Subsystem (combine closed valves, throat plug & bleed lines)	TPA-inlet
	Internal-Shaft Torque Check	Bearing-Shaft Fit	Rotate using GN_2	
	Automatic Checkout	Readiness	Comparison to Hot-Fire Data History	TPA-automatic
	External-Weldments, Ducts, Manifolds & Lines	Leak Check	Lights & Color Change Tapes or Elastomeric Paints	TPA-as installed
Flow Control Valves	External-Total Valve Inventory	Leak Check (internal)	Pressure to Verify Seal Operation	Engine as installed
	Automatic Checkout	Readiness, Actuation Timing, Position Accuracy	Comparison to Norm Data	Engine-as installed
	External-Valve Weldments	Leak Check (external)	Lights & Color Tapes or Elastomeric Paints	Engine-as installed
Gimbal Bearing Assembly	External-Bearing	Excessive Wear	Visual	Engine-as installed
Avionics Controller, Sensors & Electronics	Automatic Checkout	Readiness	Comparison to Norms	Engine-as installed

TABLE 33. INSPECTION OPERATION COSTS

Operation	Man-Hours	Special Equipment	Materials
Run & Review Trend Analysis	16	Trend Analysis Computer & Prog.	Computer Time (8 hours)
Inspect Injector & Combustion Chamber	12	Borescope, Light, Mirrors	
Inspect External Welds	24	None	None
Check Spark Igniters	0.5	None	None
Check Nozzle	5	None	None
Check Thrust Structure	5	None	None
Pressure Check System	32	None	Dry Nitrogen
Inspect Bearings	8	Borescope	None
Spin Pumps	8	Stethoscope	Dry Nitrogen
Torque Check H_2 Pump	4	Torque Wrench, Rotating Torque Measurer	None
Check Valves	12	Timer	None
Check Gimbal Bearing	4	Force Transducer	None
Check Instrumentation & Controls	40	None	None
Check Avionics	8	Programmed Check-out Set	None
Unscheduled Repairs	←	See Table 31b	→
TOTAL COST*: Double Panel - $16,577; Single Panel - $15,813			

* Equipment amortized over 800 inspections (40 engines x 20 inspections/engine)

TABLE 34. UNSCHEDULED INSPECTION REPAIR COSTS

Problem Area	Probability	Average Cost of Repairs*, dollars		P x Cost, dollars	
		Double Panel	Single Panel	Double Panel	Single Panel
Thrust Chamber and Injector **	.107	29,150	24,950	3,119	2,670
Inlet Valves	.034	6,000	6,000	204	204
Main Pumps	.126	41,100	38,600	5,179	4,864
Control Valves	.034	6,000	6,000	204	204
Ignition System	.0075	1,000	1,000	8	8
Pneumatic System	.01	1,000	1,000	10	10
Ducting	.06	100	100	6	6
Gimbal	.01	700	700	7	7
Avionics	.15	7,000	7,000	1,050	1,050
				9,787	9,023

* Based upon 33% complete replacement cost.

** Requires return to manufacturing facility for major repairs.

Pressure checks will be made by inserting inflatable throat plugs in all of the segments and pressurizing the system. Leakage will be monitored to provide a check of system seals and joints. Pressure will be applied to the engine through either the purge subsystem or separate connections.

The inspection procedure for the turbomachinery includes the internal inspection of the bearings for signs of thermal damage and wear with a flexible optic borescope, using special guides (access to both oxygen turbopump bearings and to the rear hydrogen turbopump bearing is available via speed pickup instrumentation ports). Leak check of the seals via pressurization of the system for internal flow anomalies is planned.

Turbopump assemblies will be rotated briefly at a low speed and either (1) a "measured amount" of nitrogen applied in the same manner as in the pressure check, or (2) mechanical rotation made through an access port in the turbine exhaust duct with a tool designed to act on the turbine wheel. Rotational speed and speed decay will be monitored. An audio pickup will be connected to the pumps to record possible bearing noise. Minimal internal inspection is planned because of the lack of rubbing seals, the low DN values of the machines, and the relaxed turbine blade temperature (no internal blade inspection required). The turbomachinery operating parameters, loads, and general characteristics of the engine are amenable to minimum inspection procedures.

Flow control valve inspection includes the following activities: (1) total valve inventory leakage check wherein subsystems will be pressurized to verify operation, (2) verification of valve actuation timing and position, and (3) visual inspection for weldment leakage (external inspection and mirrors, lights, and leak-sensitive tapes/paint in critical areas). There is no internal valve servicing required.

Scheduled Overhaul. The scheduled overhaul activity occurs after an accumulation of 10 in-flight hours or 300 starts, whichever occurs first. The engine overhaul program has been established similar to that of the commercial airlines. A modular overhaul approach (i.e., parts are overhauled by convenient groups, as possible) is recommended and condition monitoring will be used to determine the need for overhaul. All overhaul operations are assumed to occur at a contractor depot.

To provide a low-cost overhaul approach (maximum of 25 percent of a new engine cost), the engine components requiring overhaul at the end of 300 starts (or 10 hours duration) are minimized. Only components listed below are scheduled for overhaul.

Aerospike Engine Components Scheduled for Overhaul at 10 hours or 300 cycles

1. Chamber Segment Liners and Injector
2. Turbopump Bearings and Seals
3. Spark Plugs In Combustion Wave Generator

Overhaul operations are described in Table 35 and costs are listed in Tables 36 and 37. Engines returned to the contractor depot for overhaul are processed according to specifications for disassembly, replacement, repair, cleaning, assembly, checkout and test, and inspection requirements for each phase of the overhaul process. Engine disassembly for overhaul is directed toward chamber segments, igniters, and turbopump bearing and seals. For turbopump removal, the power pack (both turbopump assemblies and major valves) is removed vertically (Fig. 50) from within the aerospike assembly along with the conical thrust mount after frame members are detached and the welded turbine inlet lines, base bleed line, and pump discharge line are cut.

The pneumatic system (solenoid valves, relief valves, transducers, filters, etc.) is modularized and centrally mounted on the thrust structure. Once the power pack is removed, refurbishment on any particular component becomes direct and accessible (Fig. 51).

TABLE 35. OVERHAUL OPERATIONS

Operation	Procedure
1. Purge Engine	Run dry nitrogen through all flow passages
2. Remove Engine	Engage engine-carrying dolly, unbolt fluid connections and gimbal, disconnect electrical connections
3. Package Engine for Shipment	Cap all openings, place in shipping cannister, enclose trend analysis results from all inspections and current trend analysis
4. Ship to Rocketdyne	Ship engine in cannister by truck
5. Review Analysis	All indicated inspections or repairs are added to schedule
6. Dismantle Thrust Mount	Bolted Assembly
7. Remove Main Turbopumps	Cut welds to hot-gas manifold and all propellant lines
8. Replace Bearings	Dismantle pumps, change bearings, inspect parts, balance check, reassembly and test
9. Inspect Hot-Gas Ducts and Ignition Components	Visually inspect ducts, ignition components and combustors
10. Remove All Feed System Components and Actuator Brackets	Cut welds and clear combustion chamber region
11. Remove thrust Chamber Assembly	Remove upper gas manifold, cut welds at injector and at middle manifolds
12. Disassembly Thrust Chamber Assembly	Cut nozzle loose and unbolt structural rings in combustion region
13. Cut injectors from segment lines	Machine cut and inspect

TABLE 35. (Concluded)

Operation	Procedure
14. Replace Segment Lines	Reweld using same welding rig as used in manufacturing
15. Check Feed System Components	Torque and sound check pump bearings, visually inspect pumps, visually inspect valves, functional check of valves.
16. Reinstall Feed System Components and Brackets	Reweld lines
17. Remove and Replace Spark Plugs	Combustion wave ignition premixer (only)
18. Reinstall Turbopumps	Reweld into hot-gas ducts and onto lines
19. Reinstall Thrust Mount	
20. Pressure Check Engine	Seal engine and apply pressure to flow passages
21. Repack Engine	
22. Ship to Test Stand	
23. Acceptance Test Engine	Run engine over entire range, restart, check engine monitoring parameters, perform trend analysis, align gimbal bearing
24. Repack Engine and Ship to Flight Operations	

TAГ ... 36. OVERHAUL COSTS

Operation	Man-Hours	Special Equipment*	Materials
Purge Engine	2	None	Dry Nitrogen
Remove Engine	80	Dolly	None
Package Engine	40	Cannister	Desiccant
Ship Engine	-	-	-
Remove Instrumentation & Control Lines	80	-	-
Review Analysis	16	Computer Time (8 hours)	Trend Analysis Program
Remove Turbopumps	240	Translating Cutter, Tubing Cutter	-
Replace Bearings & Test	320	-	Bearings
Inspect Ducts & Preburners	16	Borescope	-
Remove All Feed System Components	160	Tube Cutter	-
Remove Combustion Chamber Segments	240	-	-
Check Injector	144	Borescope, Alignment Checker	-
Replace Combustion Chamber Segments	240	-	New Combustion Chamber
Check Components	160	-	-
Replace Components	80	-	-
Replace Spark Plugs	8	-	Spark Plugs
Replace Main Turbopumps	120	-	-
Place & Check Instrumentation & Control Lines	320	-	-
Pressure Check Engine	32	-	Dry Nitrogen
Repack Engine	40	Cannister	Desiccant
Ship to Test Stand	-	-	-
Acceptance Test	-	-	-
Repair & Ship	40	Cannister	Desiccant
Remount on Vehicle & Checkout	120	Dolly, Torque Wrenches, Test Equipment	Seals
TOTAL, SCHEDULED REPAIRS			
Unscheduled Repairs		See Table 32b	
TOTAL COST: Double Panel - $190,219; Single Panel - $175,076			

*Equipment amortized over 160 overhauls (40 engines x 4 overhauls/engine)

TABLE 37. UNSCHEDULED OVERHAUL REPAIR COSTS

Problem Area	Probability	Average Cost of Repair*, dollars		P x Cost, dollars	
		Double Panel	Single Panel	Double Panel	Single Panel
Thrust Chamber and Injector	.107	24,400	20,400	2,611	2,183
Inlet Valves	.034	6,000	6,000	204	204
Main Pumps	.126	41,100	38,600	5,179	4,864
Control Valves	.034	6,000	6,000	204	204
Ignition System	.0075	1,000	1,000	8	8
Pneumatic System	.01	1,000	1,000	10	10
Ducting	.06	100	100	6	6
Gimbal	.01	700	700	7	7
Avionics	.15	7,000	7,000	1,050	1,050
				9,279	8,536

*Based upon 33% complete replacement cost.

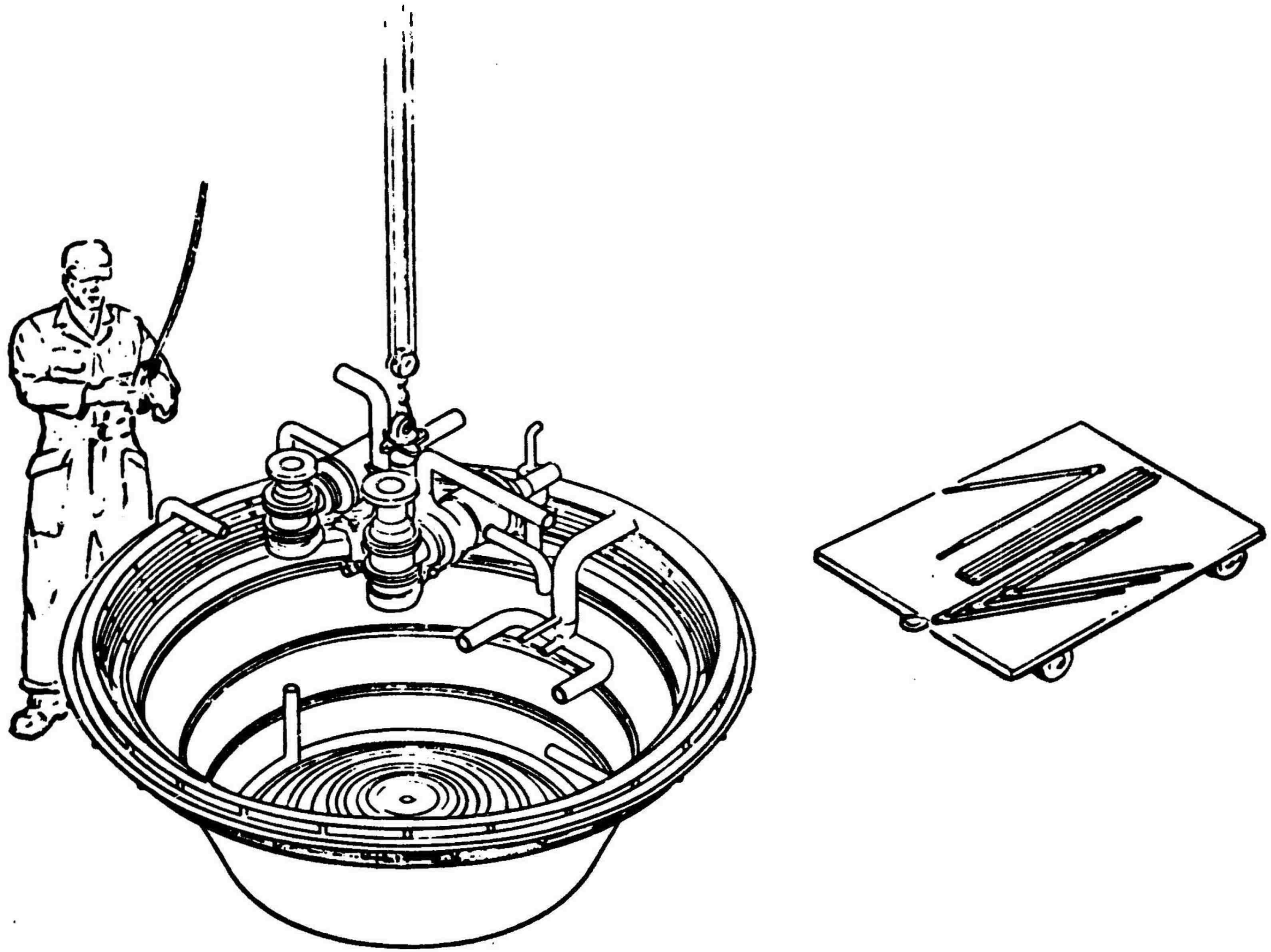


Figure 50. Power Pack Removal

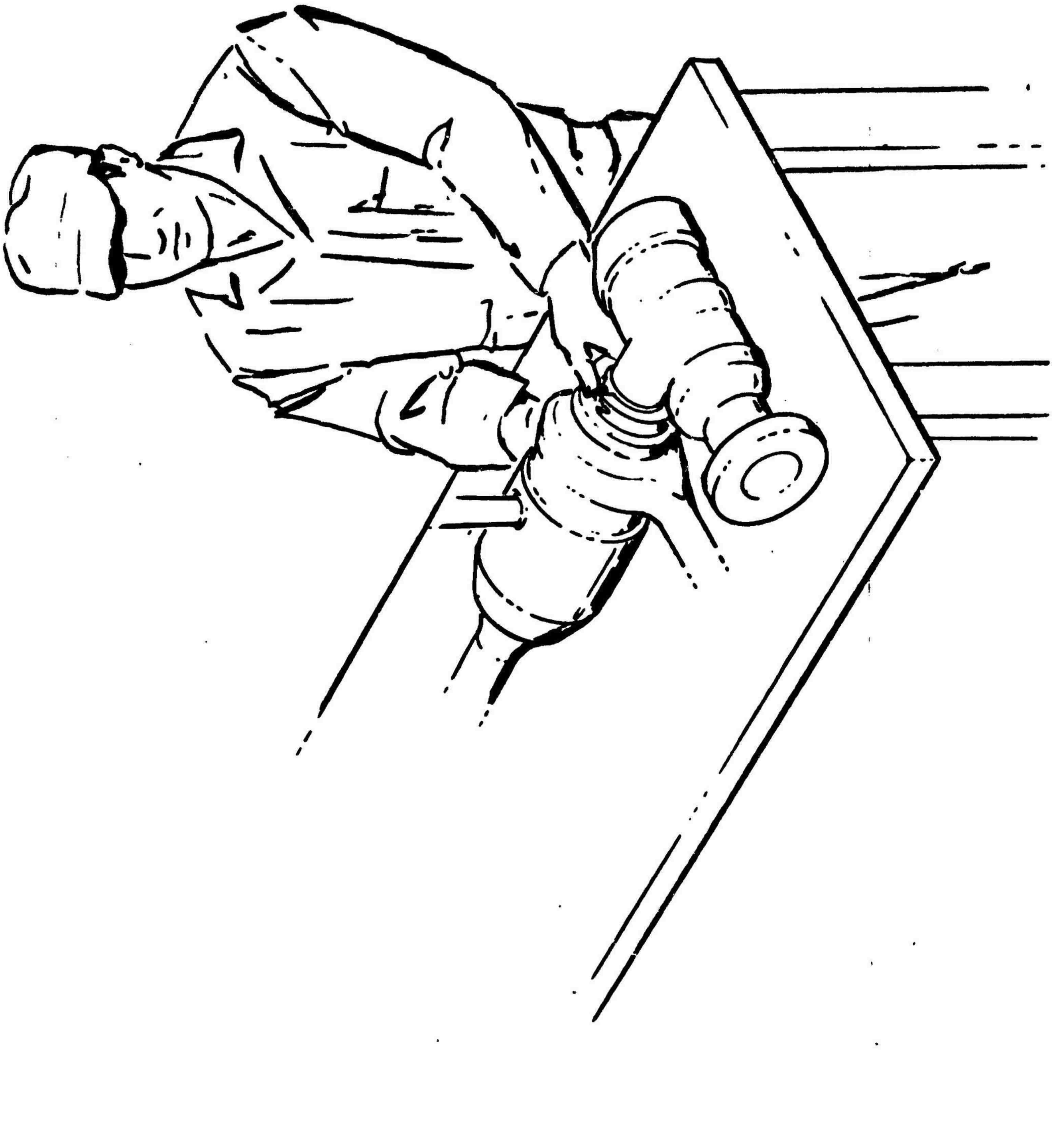


Figure 51. Main Propellant Valve Removal From Pump Assembly

For thrust chamber assembly components (except the segment subassemblies), similar actions occur where the primary nozzle and base closure can be refurbished at the engine service area. However, because segment assembly refurbishment includes complete structural ring disassembly and activities on a large number of welded lines, it is preferred that the engine be returned to the manufacturing site for completion of this type of activity. All components not specifically requiring disassembly for overhaul operations are subjected to a thorough functional checkout. If a malfunction is detected, the component is then returned to the manufacturer for disassembly and repair.

Overhauled engines receive an electrical and mechanical checkout at the overhaul facility before transportation to the engine static test facility for acceptance testing and final checkouts to verify integrity of the system. Transportation to the operational base completes the overhaul maintenance loop.

Space-Based Maintenance

Maintenance and repair of the engine in space was evaluated in light of the currently defined ground-based procedures and orbital work and repair assessments contained in Ref. 2 through 4 . Three phases of operation were considered: (1) extra-vehicular (EV) activity, (2) space station activity, and (3) ground activity. In the assessment, external installation of the engine on the vehicle was assumed.

Because of the lack of well-defined extra-vehicular (EV) repair techniques and current indications that EV work activity places strenuous demands on life support, EVA was restricted to inspection activity. No activity involving cutting or welding is conducted EV. Refurbishment activities will be conducted in a "shirt sleeve" environment within a space station. The space station is assumed to provide some positive gravity. Access to the engine during the space station activities would be accomplished through EVA removal of the complete engine and transfer to the space station. Because of the special equipment necessary for

overhaul and major repairs, refurbishment activity at the space station will consist primarily of component or module removal for unscheduled repair activity. All scheduled overhaul activity will be performed by returning the engine to a ground-based facility.

The sequence of operations associated with space-based maintenance is summarized in Fig. 52 and can be described as follows:

1. Normal Turnaround, which includes the initial launch preparations, the mission, and post-flight operations in a "go" condition, i.e., no scheduled maintenance activity is required.
2. Scheduled Inspection, which includes the Inspection and Refurbishment activity loop at the engine services area located in orbit at a space station dock whether normally scheduled or because of a "no-go" for repeat mission condition.
3. Scheduled Overhaul, which includes extensive maintenance activity at the engine facility and subsequent acceptance testing at the engine static test facility prior to return to the vehicle space operations base.

The scheduled inspection activities (Fig. 52) are routinely performed after an in-flight duration of 2 hours is accumulated or 60 thermal cycles, whichever occurs first. No space-based work is routine (scheduled) for any time interval less than this interval. In the space-based concept, the vehicle will return to an oriented operations service area (space station) after mission completion for assignment to one of the three maintenance actions paths (Fig. 52): (1) Normal Turnaround, (2) Scheduled Inspection, or (3) Scheduled Overhaul.

Overhaul is scheduled to occur (on the ground at an engine facility) after a duration of 10 in-flight hours is accumulated or 300 thermal cycles, whichever

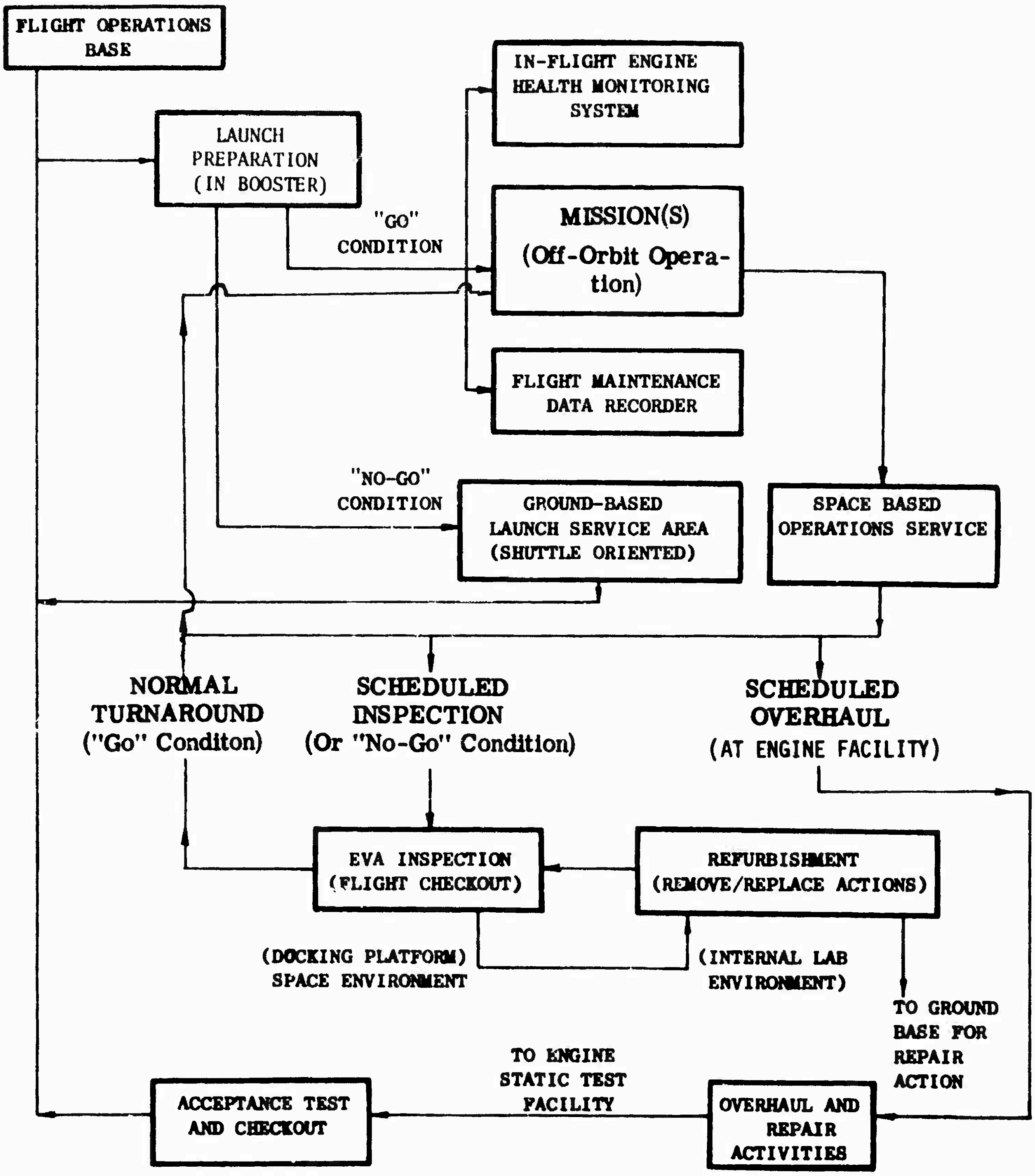


Figure 52. Space-Based Engine Maintenance Plan Sequence

occurs first. Overhauled engines will meet all new engine requirements and will be subjected to component and hot engine acceptance testing (at the engine static test facility) identical to a new engine. The overhaul activity will be conducted at an engine facility and its cost will not exceed 25 percent of a new engine cost. Installation in the vehicle will occur in orbit.

Scheduled Inspection. The scheduled inspection is performed in space and is begun with a complete flight-type automatic checkout as the first step. This checkout is conducted using the same vehicle systems (avionics package) that are used for automatic checkout prestart in the space environment. The engine measurements that are monitored are listed in Table 38. These checkout results, analyzed by the on-board computer, are provided to the maintenance crew and to the trend analysis computer program (which may be ground based). In the space-based inspection activity, the analysis of these results will assume a greater role in making refurbishment decisions.

The EV Inspection procedure involves the engine as installed in the vehicle and occurs in a dock area external to the space station. Because the base closure is nonremovable, all inspection activity occurs from the top side of the installation. Engine inspection is facilitated by full gimbal of the engine which provides access space from the top side of the engine while the vehicle is docked. Engine removal from the vehicle is necessary only when inspection procedures indicate the need for component removal and will occur in an EVA mode. The actions of the Inspection block (Fig. 52) are listed for major subsystems in Table 38, and are very similar to those for ground-based inspection.

The engine is inspected vi--ally (externally) for signs of structural damage or deterioration, overheating or erosion, and for external wear due to contact with adjacent parts. Anomalies such as minor fluids leakage, structural failure by cracks or deformation, local erosion in key combustion device components, externally caused damage, and foreign material or contamination also

TABLE 38. ENGINE INSPECTION PROVISIONS (Major Subsystems)

Inspection Area	Type of Inspection	Type of Fault	Inspection Technique	Access
Thrust Chamber Assembly (TCA)	Internal-Segment Liner & Injector Face	Signs of Thermal Damage (erosion, cracking, plugging)	Flexible Fiber Optic Borescope Augmented w/ Mirrors & Lights	TCA-Throat Gap
	External-Weldments, Ducts, Manifolds & Lines	Leak Check	Lights & Color Change Tapes or Elastomeric Paints (Hand-Held Mass Spectrometer Possibility)	TCA-as installed
	External-Bolt Torque Check	Structural Integrity	Torque Wrench on Back-up Ring Bolts	TCA-as installed
	Internal-Spark Ignition (automatic checkout)	Readiness	Provide Current to Igniters	TCA-automatic
	Automatic Checkout	Readiness	Comparison to Hot-Fire Data History	TCA-automatic
	External-Primary Nozzle & Base Closure	Tube Cracks, Splits, Holes	Visual, Mirrors & Lights	TCA-as installed
	External-Thrust Structure	Deformation & Structural Integrity	Bolt Torque Check, Tolerance Measurement	TCA-as installed
	External-TCA	System Leak Check	Plug Throat & Pressurize Subsystems	TCA-as installed
Turbopump Assemblies (TPA)	Internal-Bearings	Signs of Thermal Damage, Spalling	Flexible Fiber Optic Borescope w/Special Guides; Use of Audio-Fault Detection Also Possible	TPA-speed. Speed pickup instrn. port

TABLE 38. (Concluded)

Inspection Area	Type of Inspection	Type of Fault	Inspection Technique	Access
Turbopump Assemblies (TPA) (cont'd)	Internal-Seals	Leak Check (functional) & Readiness (automatic)	Pres -e Subsystem (combine closed valves, throat plug & bleed lines)	TPA-inlet
	Internal-Shaft Torque Check	Bearing-Shaft Fit	Rotate using GN_2	
	Automatic Checkout	Readiness	Comparison to Hot-Fire Data History	TPA-automatic
	External-Weldments, Ducts, Manifolds & Lines	Leak Check	Lights & Color Change Tapes or Elastomeric Paints	TPA-as installed
Flow Control Valves	External-Total Valve Inventory	Leak Check (internal)	Pressure to Verify Seal Operation	Engine as installed
	Automatic Checkout	Readiness, Actuation Timing, Position Accuracy	Comparison to Norm Data	Engine-as installed
	External-Valve Weldments	Leak Check (external)	Lights & Color Tapes or Elastomeric Paints	Engine-as installed
Gimbal Bearing Assembly	External-Bearing	Excessive Wear	Visual	Engine-as installed
Avionics Controller, Sensors & Electronics	Automatic Checkout	Readiness	Comparison to Norms	Engine-as installed

are checked. Specific areas of inspection may be indicated by the automatic checkout.

Handholds, footholds, and tether attachment points are provided on the engine to provide personnel stabilization during inspection. Because the shadow side of the engine will be quite dark, a portable light unit with appropriate mounting provision will be necessary. To minimize the time spent in visual inspection and maintain an inspection record, photographs will be made for review after the EV inspection. A standard series of photographs will be made and compared to previous photographs of the same area.

For turbopump torque checks in space, the turbopumps will be rotated briefly at low speed by a "measured" amount of nitrogen and both the peak speed and speed decay monitored. An audio record of possible bearing noise also will be made. Access to the turbines for mechanical rotation would necessitate bolted access ports in the high-pressure duct which are undersirable because they would be a leakage source. The remainder of the inspection (pressure check and internal inspection) will be much like the ground-based inspection, except that all tools and checkout equipment will be of special design (Table 39).

TABLE 39. TOOLS FOR EV INSPECTION (TYPICAL)

Camera
Portable Lights
Pneumatic Supply
Portable Mass Spectrometer
Borescope Viewing Methods
Reactionless Wrenches
Reactionless Torque Measuring Device

Refurbishment. Should the inspection identify a problem, two courses of action are open: (1) removal and replacement of minor items as an EV activity, or (2) removal and replacement of the engine. The first area is restricted to small mechanically connected parts. All other work necessitates engine removal. Engine removal and replacement is facilitated by use of only mechanical connections between engine and vehicle. On removal, the engine is either taken to the space station for minor refurbishment or returned to the ground for more extensive work. In the space station, line-welded components can be removed and replaced as long as physical balancing or calibration, which require facilities, is unnecessary. Component repair and component replacement necessitating balancing or calibration is conducted on the ground.

Scheduled Overhaul. All scheduled overhaul is conducted on the ground as described in the previous section.

COMPONENT DESIGN AND ANALYSIS

THRUST CHAMBER ASSEMBLY

The aerospike thrust chamber assembly will consist of the following major components:

1. Thrust Chamber
2. Nozzle Extension
3. Base Closure
4. Thrust Mount and Gimbal Structure

Two preliminary design and development approaches are being utilized to provide a high-performing thrust chamber assembly for the aerospike engine system. The two design approaches differ specifically in the technique used to regeneratively cool the thrust chamber and have the following characteristics:

1. Double-panel, regeneratively cooled thrust chamber, $\epsilon = 200$, maximum chamber pressure = 1000 psia, where the hot-gas wall of the thrust chamber is cooled with both hydrogen and oxygen.
2. Single-panel, regeneratively cooled thrust chamber, $\epsilon = 110$, maximum chamber pressure = 750 psia, where the hot-gas wall of the thrust chamber is regeneratively cooled with hydrogen only.

A typical thrust chamber assembly is shown in Fig. 53. The major component configuration of each thrust chamber assembly type is described below.

Thrust Chamber

The thrust chamber consists of an injector and a regeneratively cooled combustion chamber (combustor) that has a low expansion ratio divergent section

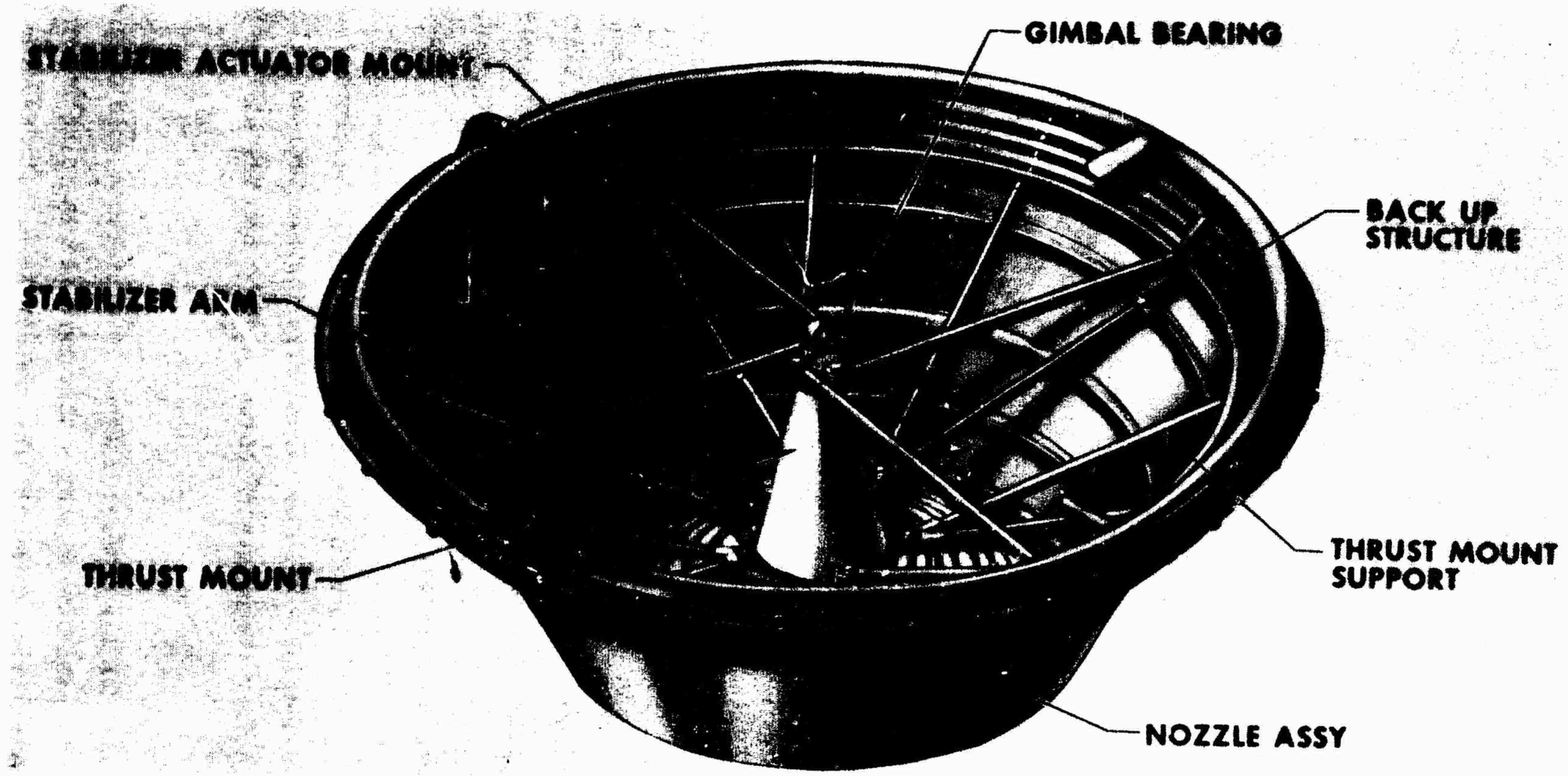


Figure 53. Aerospike Thrust Chamber Assembly

and two external structural support rings. A segmented chamber ap·roach is utilized in which 24 thrust chamber segments are stacked on a continuous inner structural ring and a continuous outer structural ring, resulting in a 360-degree circular assembly of combustors mechanically sandwiched between structural rings. In the unpressurized area at each interface between combustors, baffle, or sideplate region, bolts are installed to unite the inner and outer structural rings.

This design approach illustrated in Fig. 54, achieves an aerospike thrust chamber without bonding coolant panels to the pressure and thrust restraining structure, thus avoiding the processing associated with brazing. The resulting mechanical assembly allows removal and replacement of individual segments. A preliminary drawing of the double-panel thrust chamber is shown in Fig. 55, and the single-panel thrust chamber in Fig. 56. The combustor and injector design criteria applicable to each thrust chamber design approach are as follows.

Combustor, Double-Panel. The combustor is assembled from a basic, two-piece (split), NARloy investment casting. A two-piece, split casting is required due to an inability to reliably internally core the casting mold to obtain a 0.085-inch throat width. The coolant passage closeout procedures are slightly different from the single-panel combustor due to the use of the oxidizer for secondary cooling. The outer body has a brazed on NARloy closeout sheet the same as the single-panel chamber, but the inner body utilizes an individual tube closeout for each coolant passage, as shown in Fig. 57.

The tubes are NARloy, 0.055-inch OD with a 0.010-inch wall thickness, and formed to match the chamber contour. The tube transports the oxidizer which is used to extract heat both from the hydrogen coolant and the hot-gas wall as shown in Fig. 58.

A casting was selected for the combuster liner in preference to a machined part because of lower cost. The same criteria that resulted in selection of a non-tubular wall, NARloy material combustor for the single-panel combustor apply.

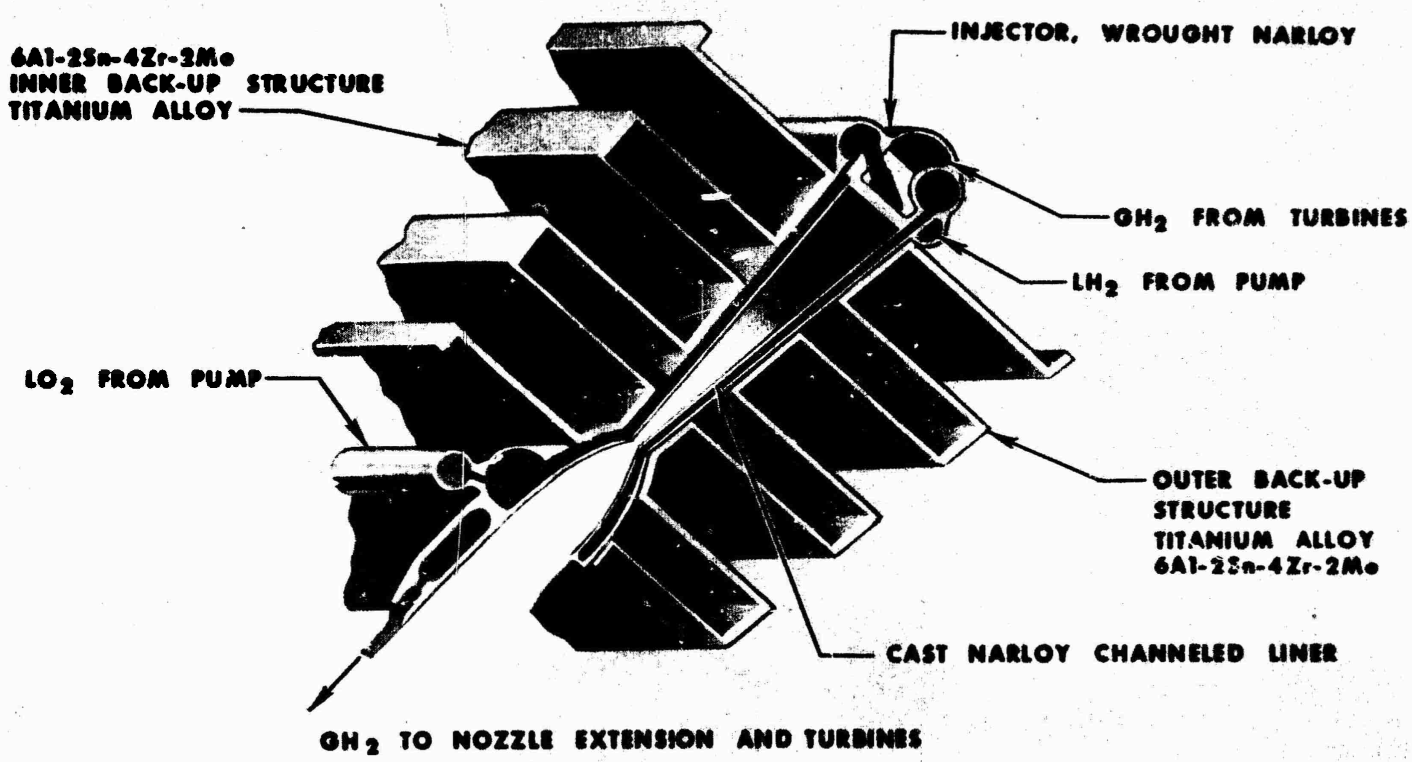
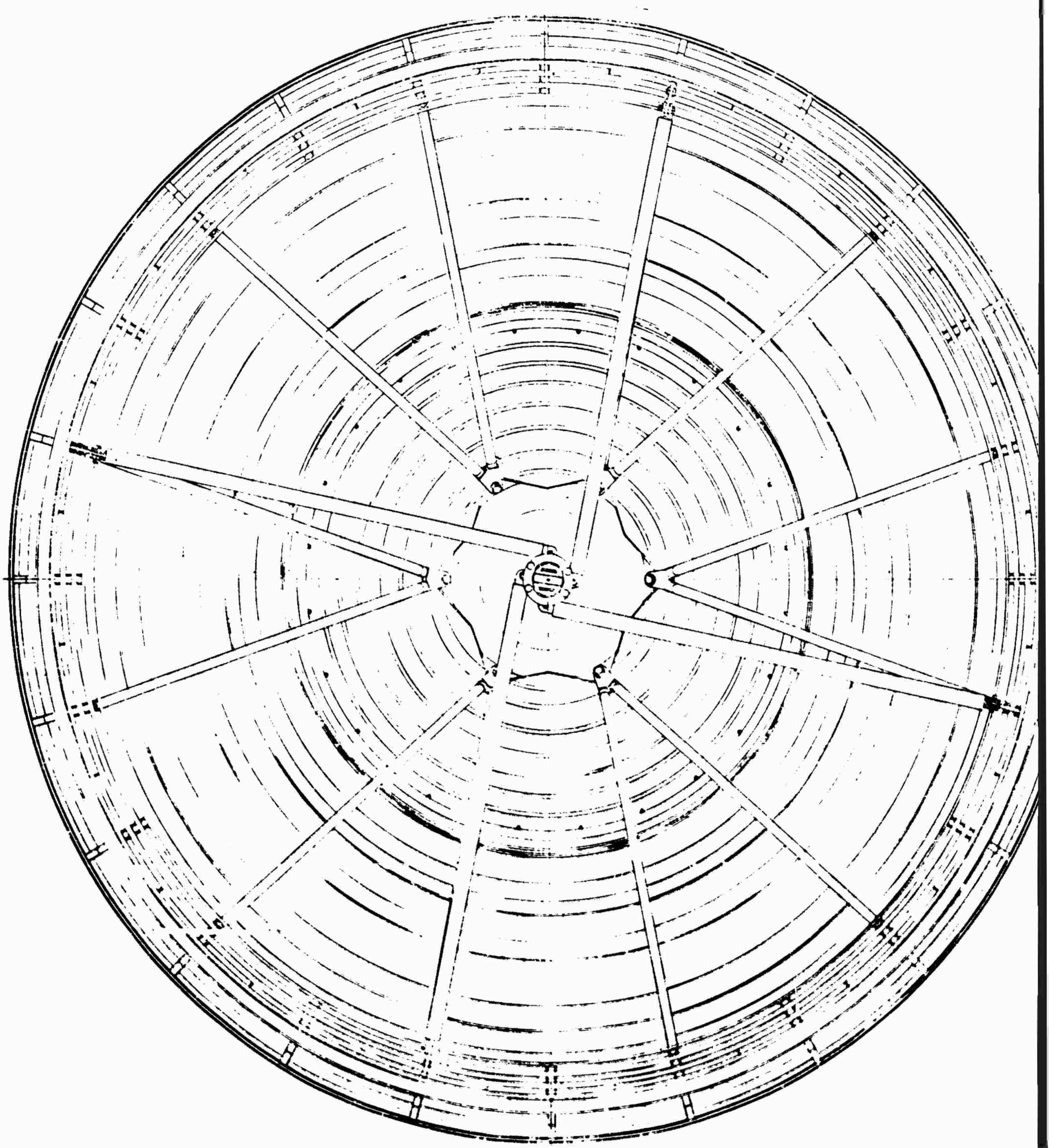
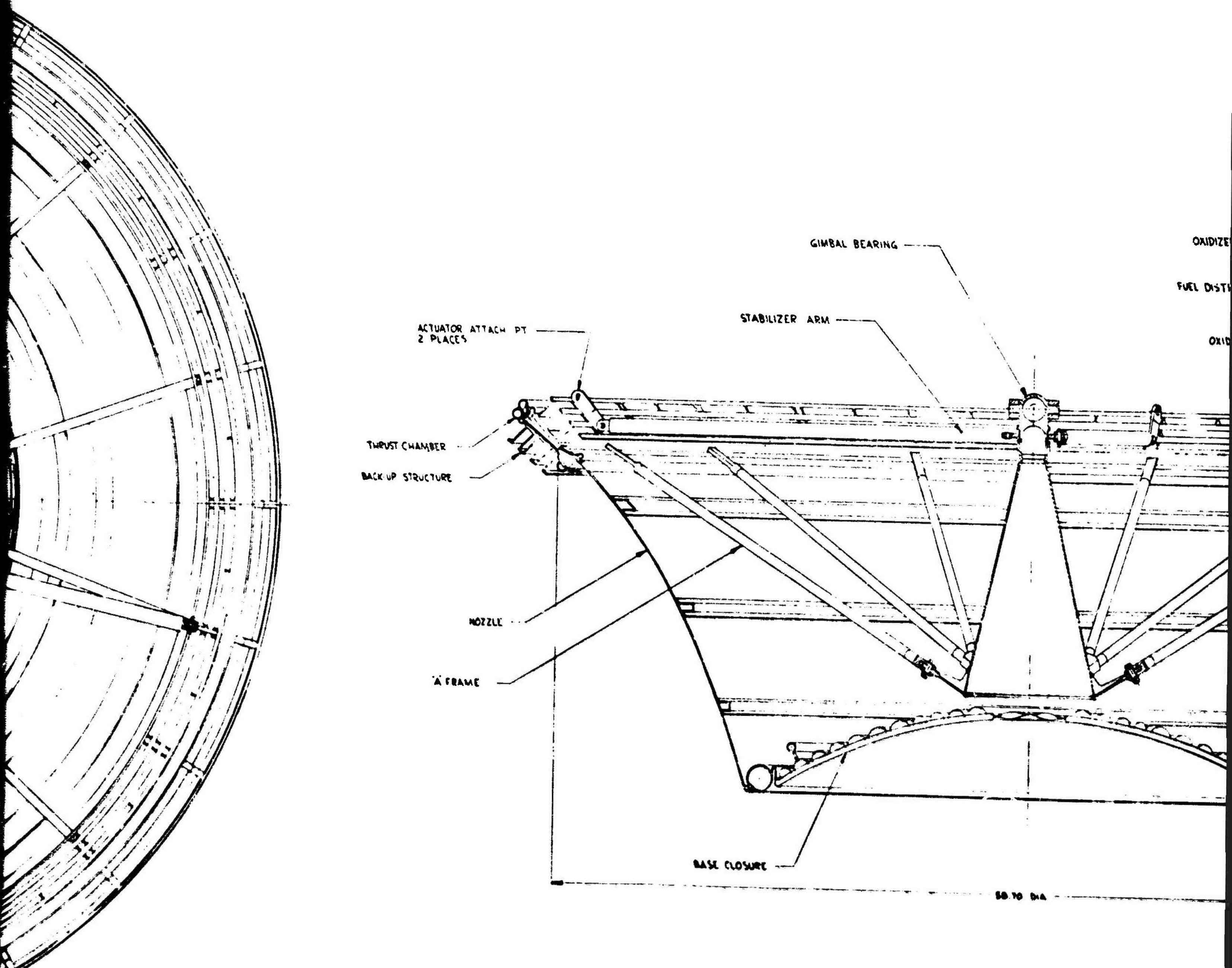


Figure 54. Aerospike Combustion Chamber Design Approach




GIMBAL BEARING
STABILIZER ARM
ACTUATOR ATTACH PT
2 PLACES
THRUST CHAMBER
BACK-UP STRUCTURE
NOZZLE
"A" FRAME
BASE CLOSURE

2

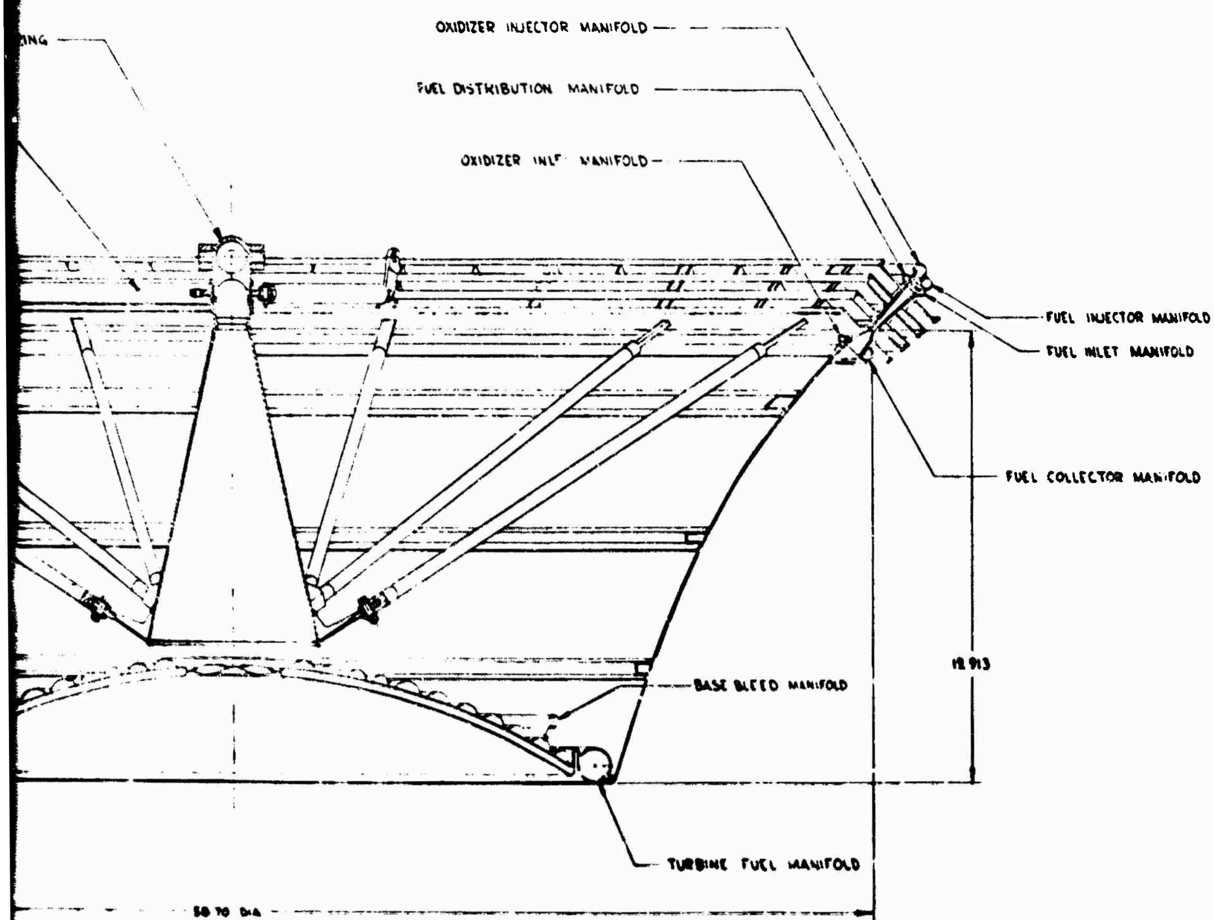
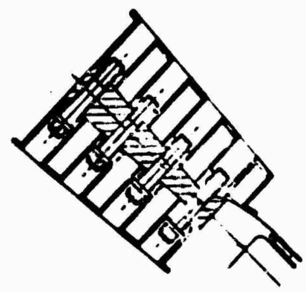


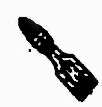
Figure 55. Thrust Chamber Assembly; Double-Panel Aerospike

3

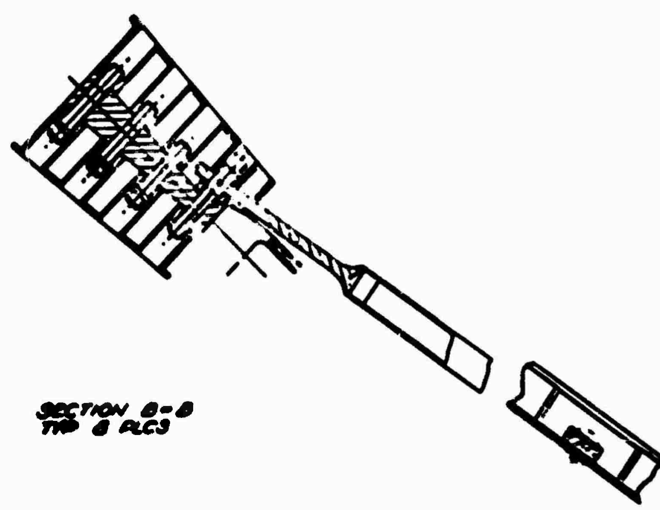


SECTION C-C
TYP 12 PLCS

SECT D-D

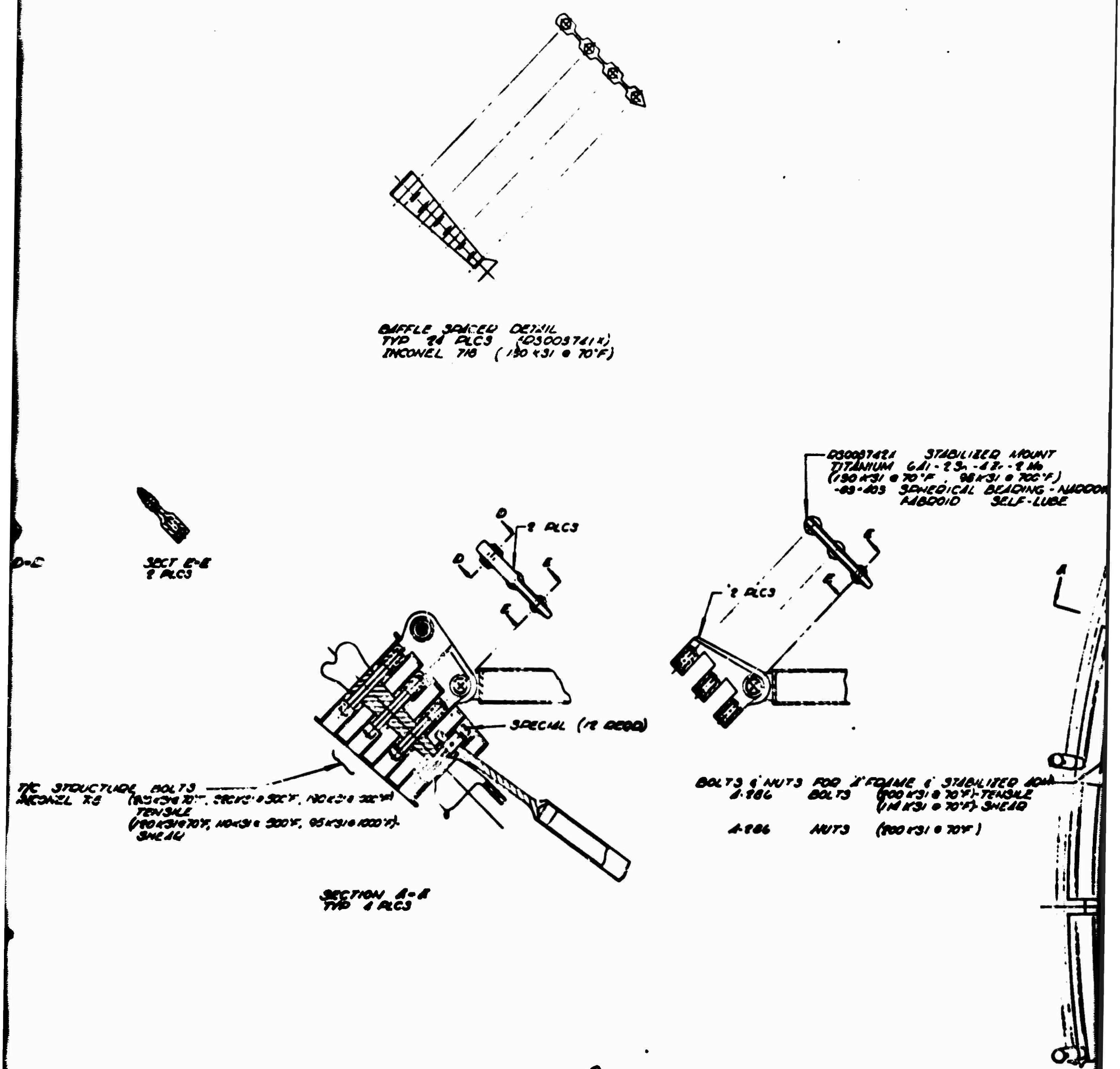


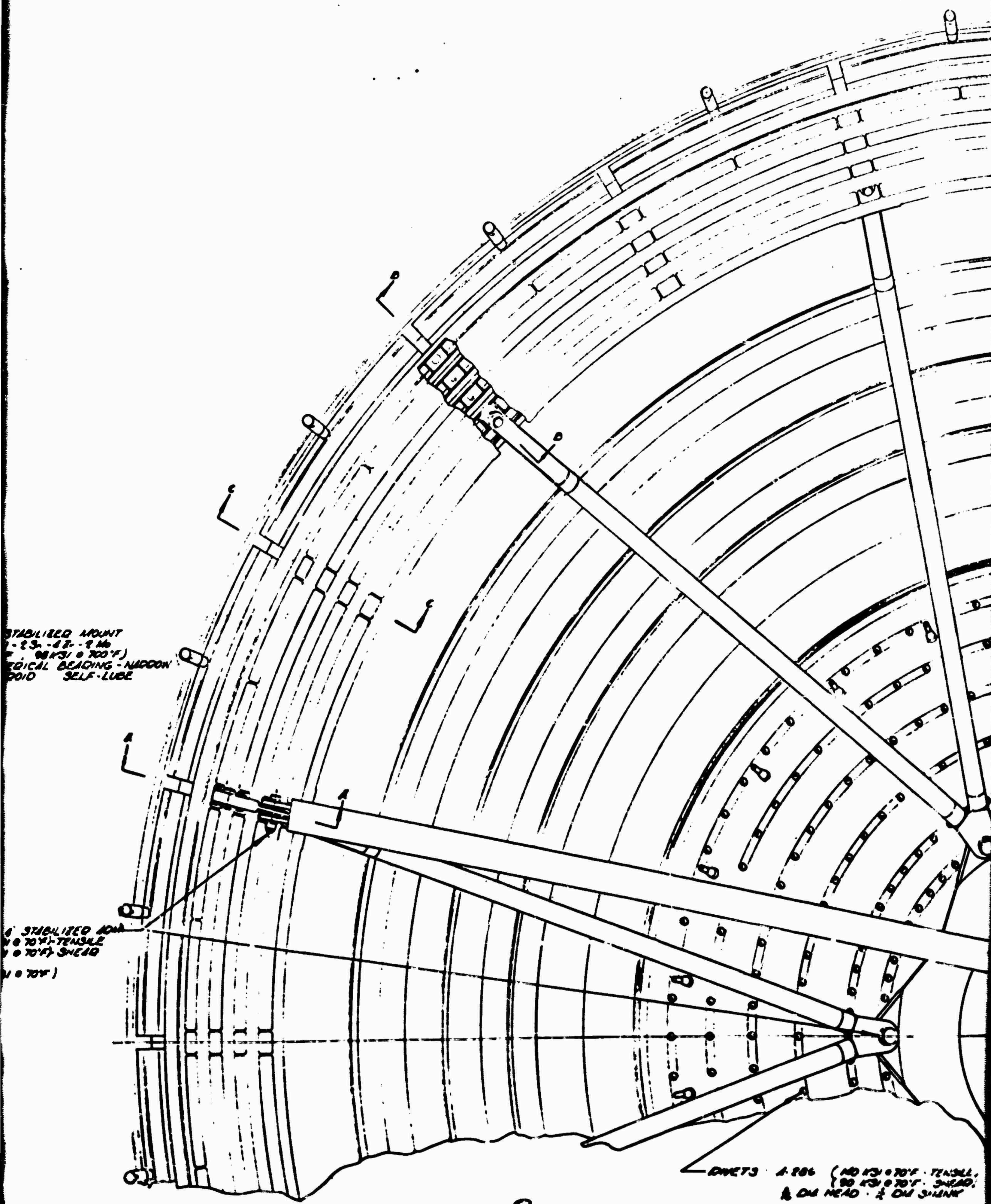
SECT E-E
8 PLCS

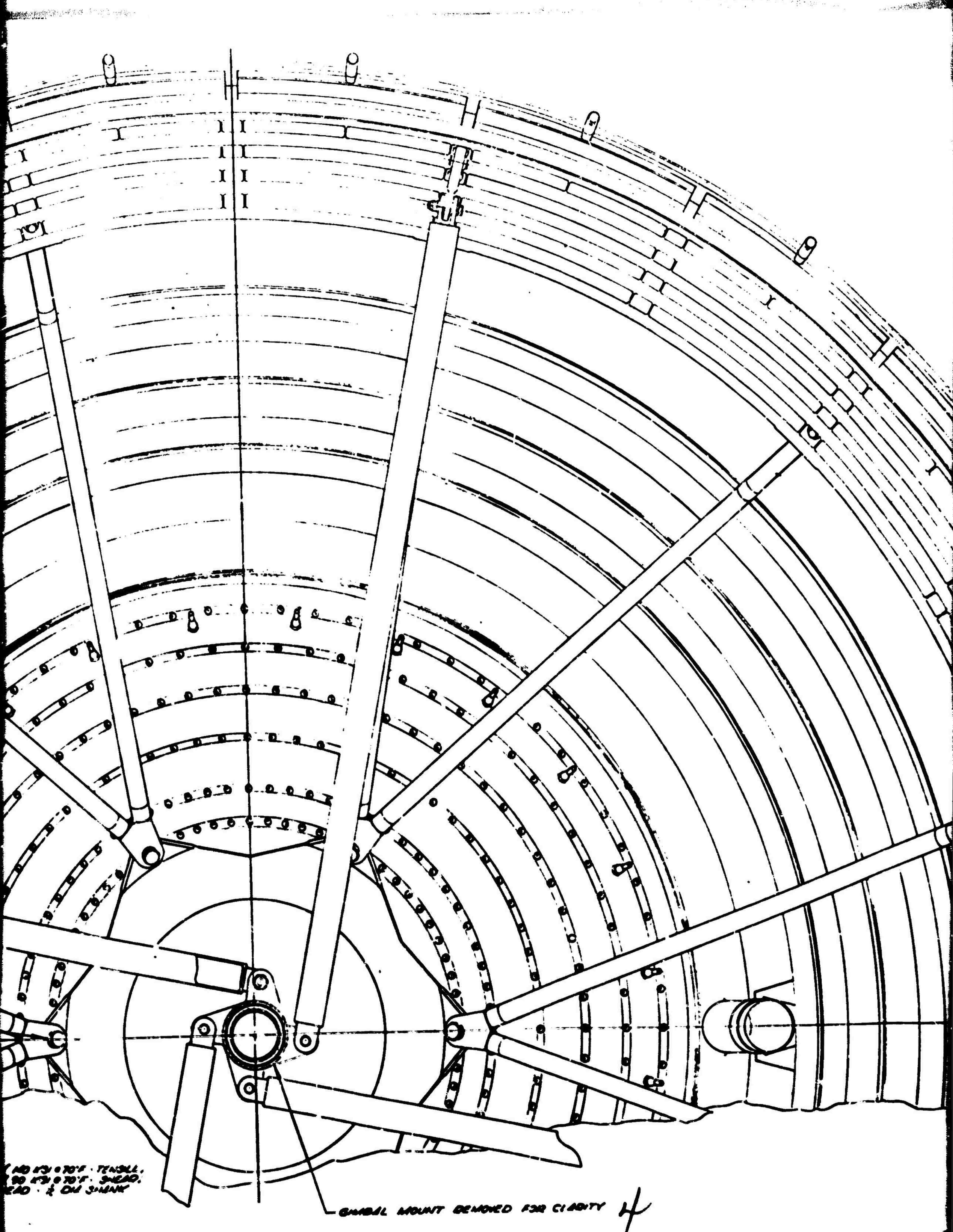


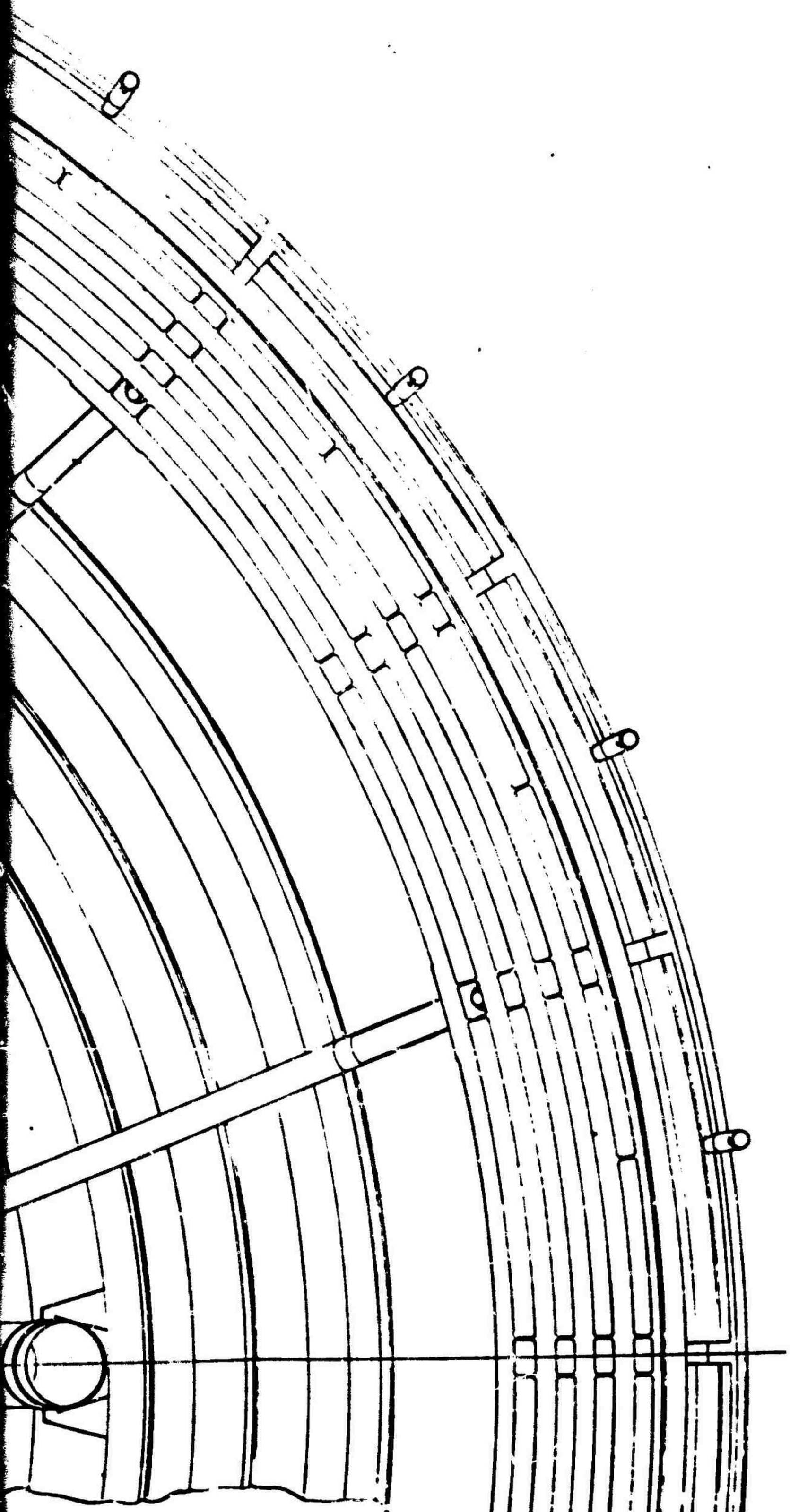
T/C STRUCTURAL BOLTS
INCONEL X-8
TENSILE
SHEAR

SECTION B-B
TYP 8 PLCS


BAFFLE SPACER DETAIL
TYP 24 PLCS
INCONEL 718 (190 KSI @ 70°F)
SECT E-E
2 PLCS
2 PLCS
SPECIAL (12 REQD)
T/C STRUCTURE BOLTS
TENSILE
SHEAR
SECTION A-A
TYP 4 PLCS
STABILIZED MOUNT
TITANIUM
SPHERICAL BEARING
SELF-LUBE
2 PLCS
BOLTS & NUTS FOR
BOLTS
TENSILE
SHEAR
A-286 NUTS


STABILIZED MOUNT
(98 KSI @ 700°F)
BEARING
SELF-LUBE
STABILIZED
@ 70°F) - TENSILE
@ 70°F) SHEAR
@ 70°F)
RIVETS A-286
DIA HEAD
DIA SHANK


KSI @ 70°F · TENSILE.
KSI @ 70°F · SHEAR.
½ DIA SHANK
GIMBAL MOUNT REMOVED FOR CLARITY
4



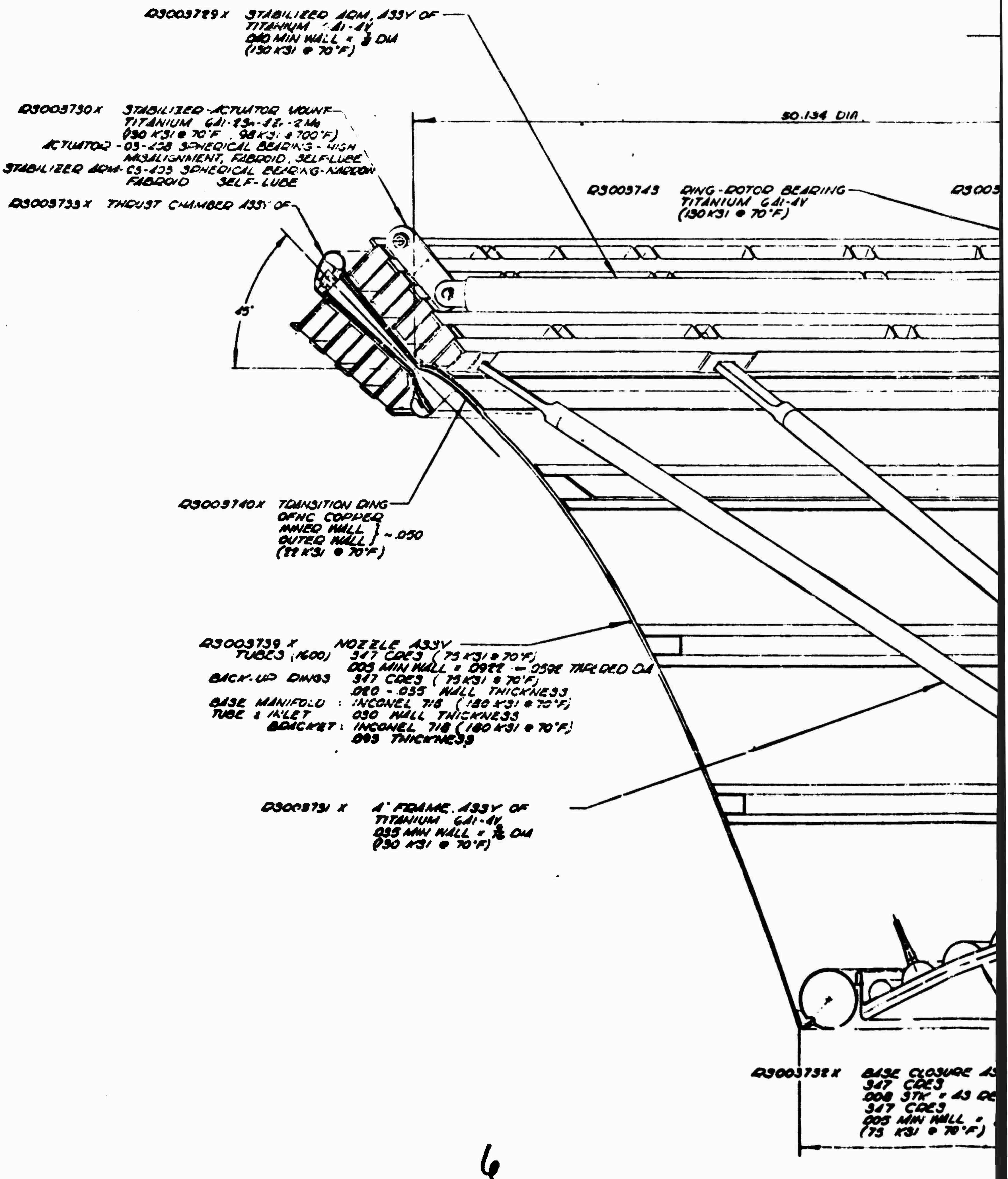
Q300

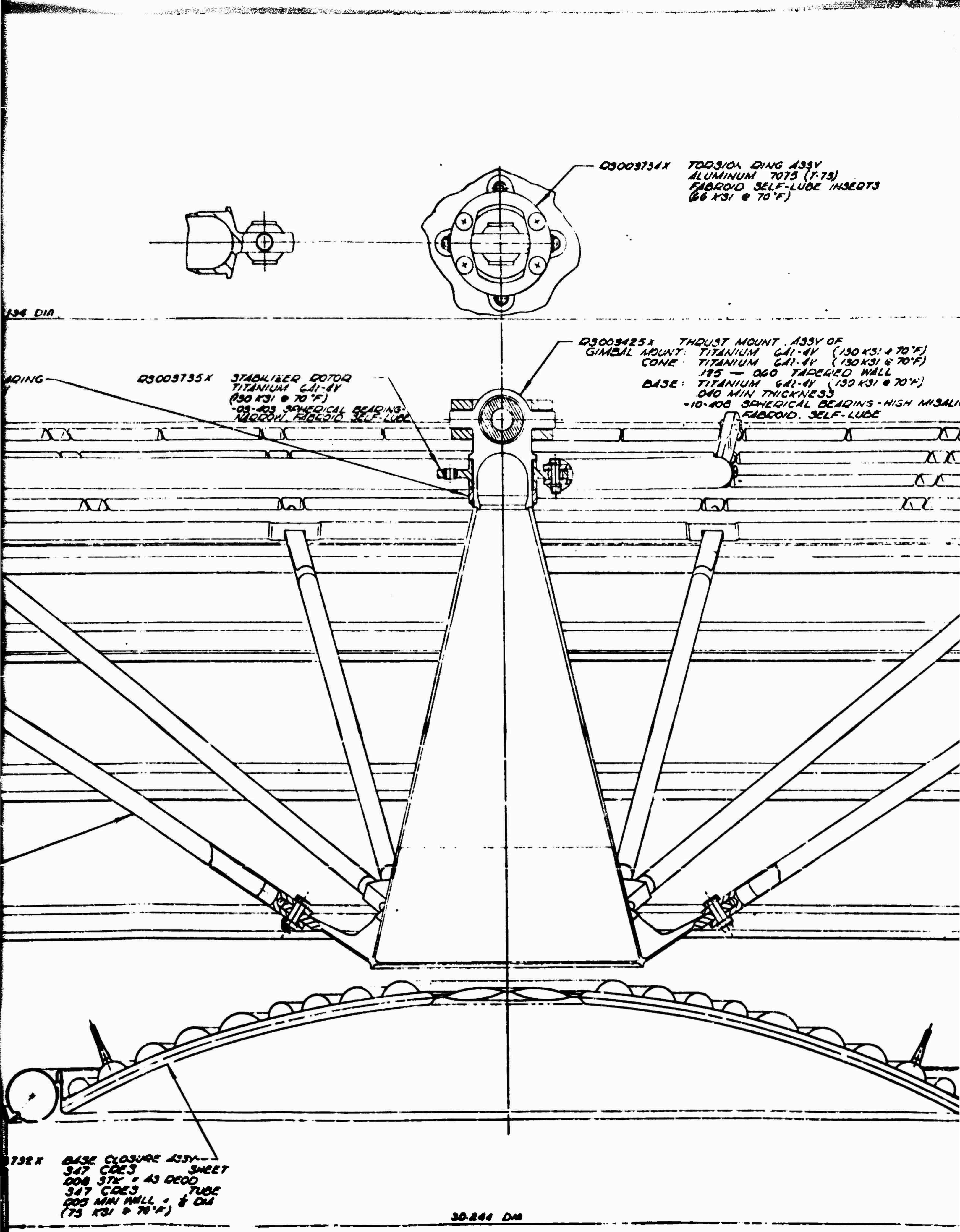
AC

STABILIZ

Q30037

5


R3003729X STABILIZER ARM, ASSY OF
TITANIUM 6Al-4V
.040 MIN WALL x ¾ DIA
(130 KSI @ 70°F)
R3003730X STABILIZER-ACTUATOR MOUNT
TITANIUM 6Al-2Sn-4Zr-2Mo
(130 KSI @ 70°F, 98 KSI @ 700°F)
ACTUATOR - CS-438 SPHERICAL BEARING - HIGH MISALIGNMENT, FABROID, SELF-LUBE
STABILIZER ARM - CS-433 SPHERICAL BEARING - NARROW FABROID SELF-LUBE
R3003733X THRUST CHAMBER ASSY OF
45°
50.134 DIA
R3003743 RING-ROTOR BEARING
TITANIUM 6Al-4V
(130 KSI @ 70°F)
R3003740X TRANSITION RING
OFHC COPPER
INNER WALL
OUTER WALL } ~.050
(22 KSI @ 70°F)
R3003739X NOZZLE ASSY
TUBES (1600) 347 CRES (75 KSI @ 70°F)
.005 MIN WALL x .0922 - .3592 TAPERED DIA
BACK-UP RINGS 347 CRES (75 KSI @ 70°F)
.020 - .035 WALL THICKNESS
BASE MANIFOLD : INCONEL 718 (180 KSI @ 70°F)
TUBE & INLET .030 WALL THICKNESS
BRACKET : INCONEL 718 (180 KSI @ 70°F)
.093 THICKNESS
R3003731X 4" FRAME, ASSY OF
TITANIUM 6Al-4V
.035 MIN WALL x ⅞ DIA
(130 KSI @ 70°F)
R3003732X BASE CLOSURE AS
347 CRES
.008 STK x 43 RE
347 CRES
.005 MIN WALL x
(75 KSI @ 70°F)



D3003734X TORSION RING ASSY
ALUMINUM 7075 (T-73)
FABROID SELF-LUBE INSERTS
(66 KSI @ 70°F)
.134 DIA
D3003425X THRUST MOUNT . ASSY OF
GIMBAL MOUNT: TITANIUM 6Al-4V (130 KSI @ 70°F)
CONE: TITANIUM 6Al-4V (130 KSI @ 70°F)
.125 → .060 TAPERED WALL
BASE: TITANIUM 6Al-4V (130 KSI @ 70°F)
.040 MIN THICKNESS
-10-408 SPHERICAL BEARINGS - HIGH MISALI
FABROID, SELF-LUBE
ARING
D3003735X STABILIZER ROTOR
TITANIUM 6Al-4V
(130 KSI @ 70°F)
-03-403 SPHERICAL BEARING-
NARROW, FABROID SELF-LUBE
732X BASE CLOSURE ASSY
347 CRES SHEET
.008 STK × 43 REQD
347 CRES TUBE
.005 MIN WALL × ⅛ DIA
(75 KSI @ 70°F)
30.244 DIA

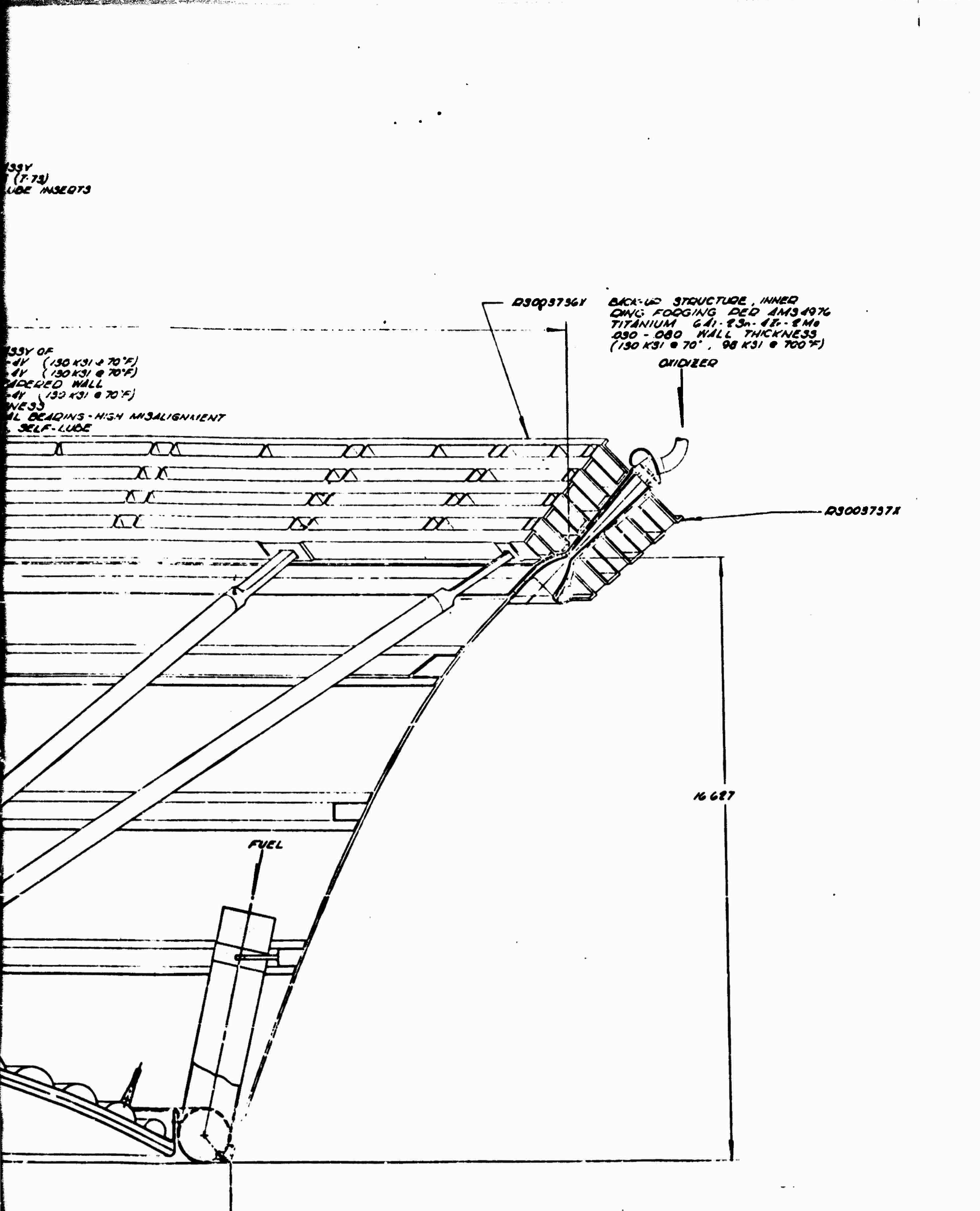


Figure 56. Thrust Chamber Assembly; Single-Panel Aerospike

8

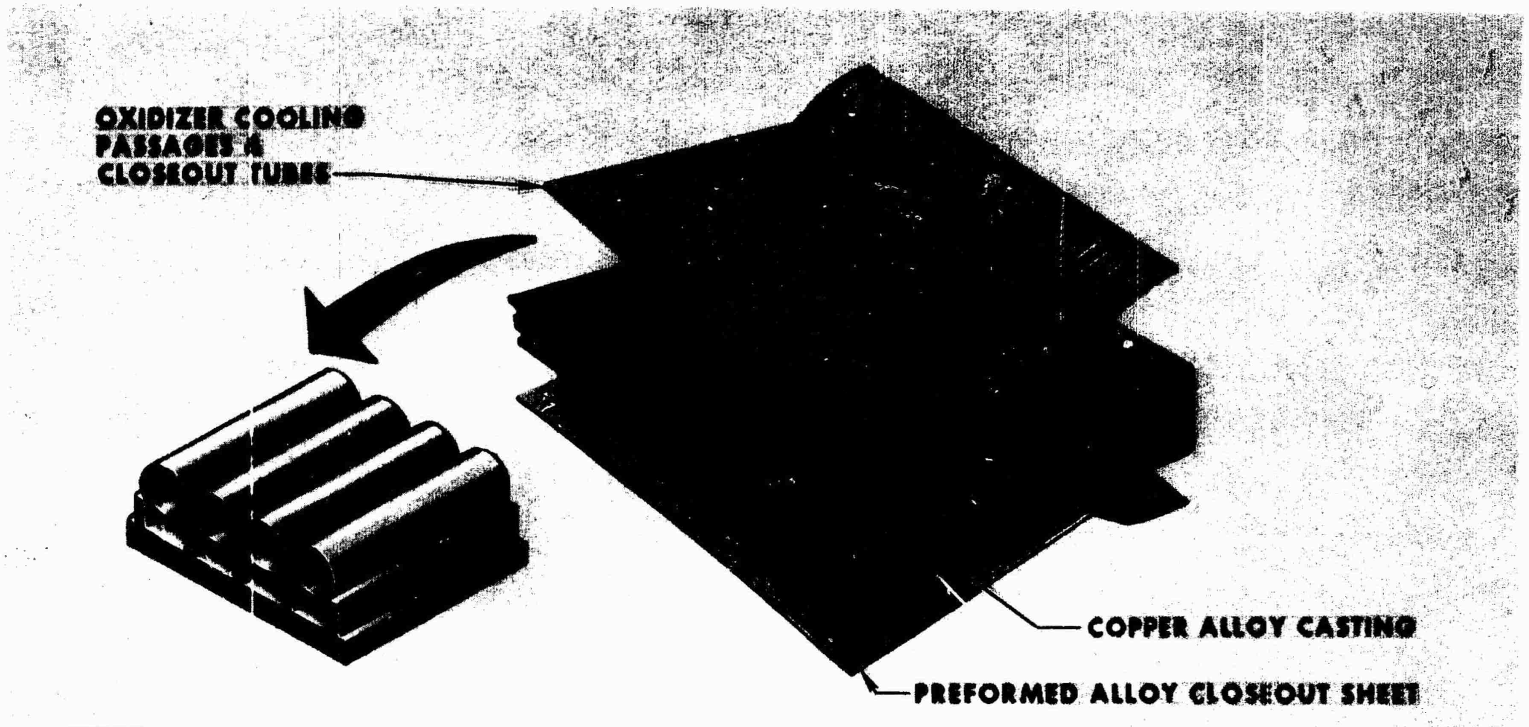


Figure 57. Double-Panel Thrust Chamber Segment

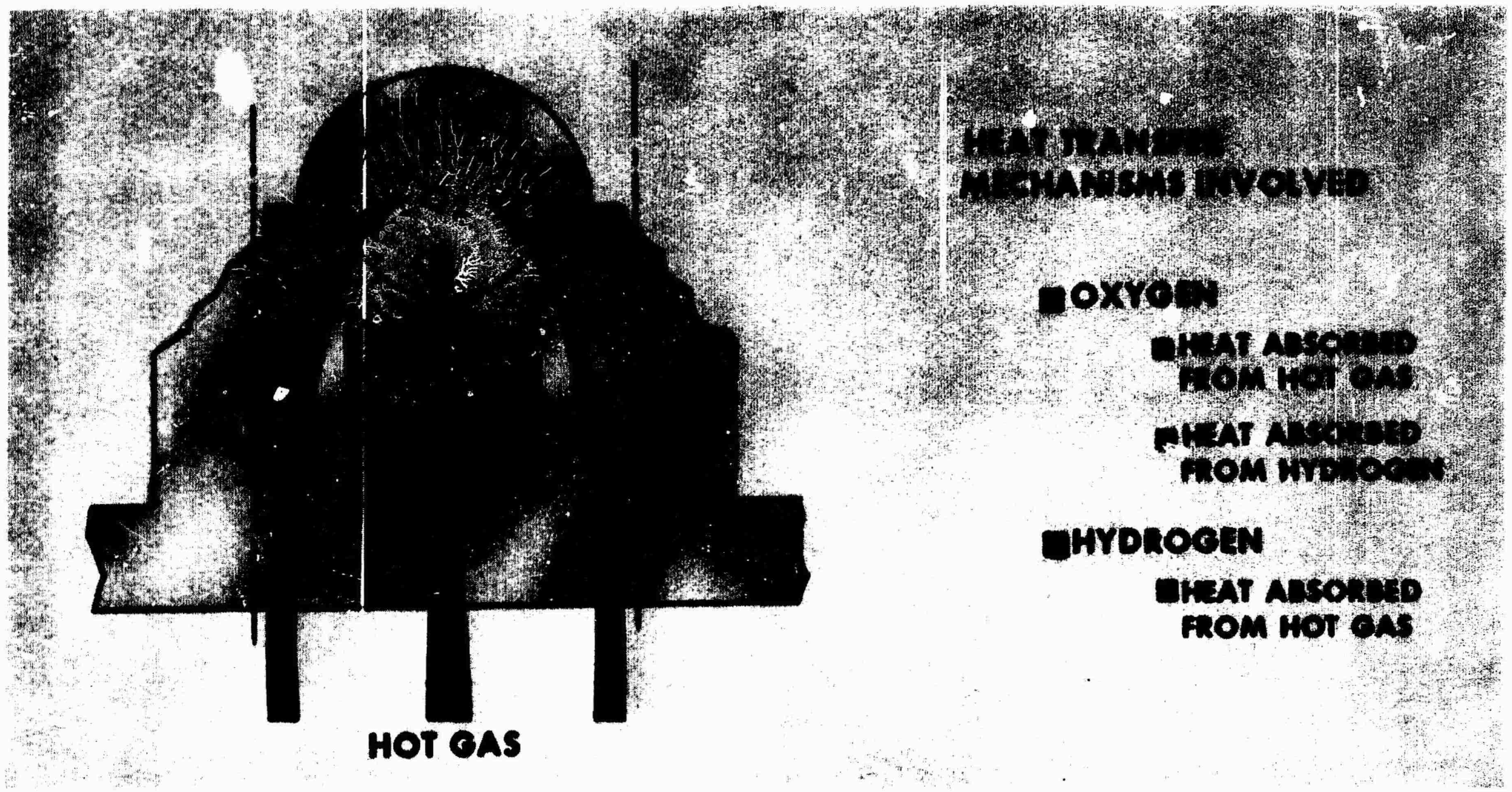


Figure 58. Double-Panel Cooling Concept

The cooling circuit selected for the combustor is shown in Fig. 59, and consists of a double-pass, combustor-first, nozzle-last type of circuit. The hydrogen coolant enters the outer body first and completes an up and down traverse followed by a down-pass through the side panels, up- and down- pass through the inner body, and completes the circuit by flowing single-pass down through the nozzle.

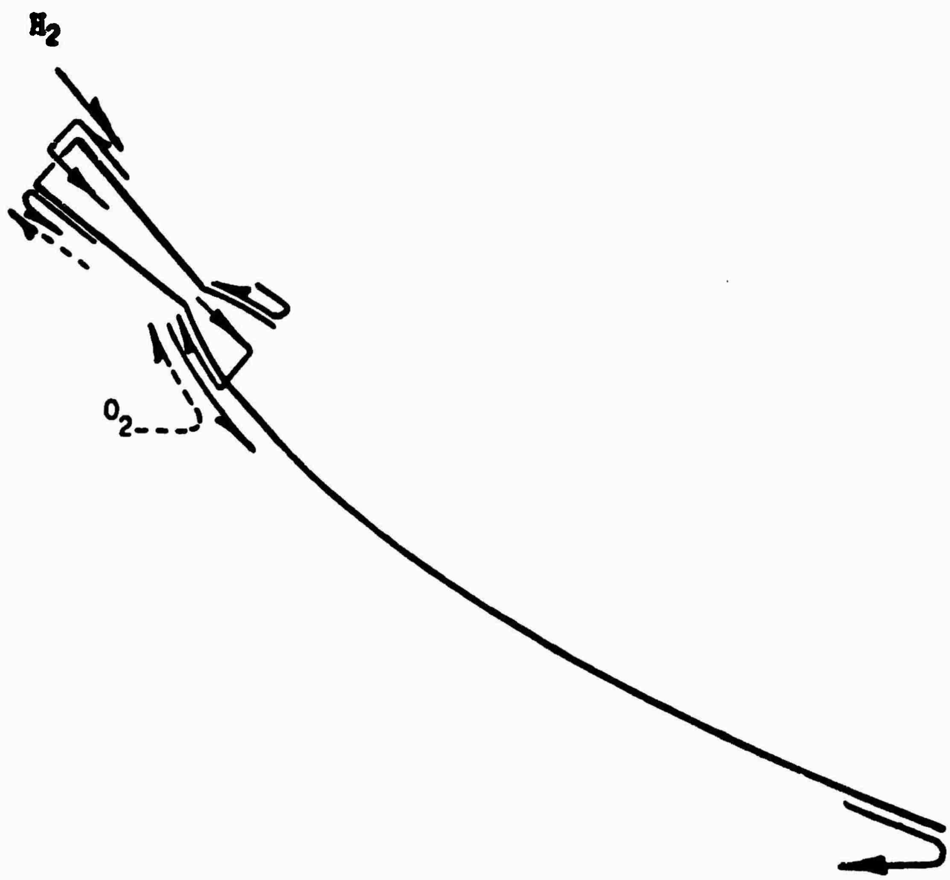


Figure 59. Double-Panel Demonstrator Thrust Chamber Cooling Circuit

The oxidizer, which is used for secondary cooling, completes a single up-pass circuit through the tubes that are attached to the inner body.

Preliminary analysis was based on extrapolated experimental hot-gas side heat transfer film coefficients, shown in Fig. 60. A double-pass coolant circuit, down and up for adjacent coolant passages, was required because of the higher design point chamber pressure, 1000 psia versus 750 psia for the single panel which could be cooled with a single-pass circuit.

Sizing of the coolant passages was completed through use of the extrapolated heat transfer coefficient profile, Fig. 60. Channel and oxidizer tube dimensions at the minimum flow area are:

<u>Inner Body (90 Channels)</u>
Channel Width = 0.040 inch
Channel Depth = 0.025 inch
Land Width = 0.035 inch

<u>Side Panel (2 Channels)</u>
Channel Width = 0.06 inch
Channel Depth = 0.15 inch
Land Width = 0.040 inch

<u>Outer Body (90 Channels)</u>
Channel Width = 0.040 inch
Channel Depth = 0.020 inch
Land Width = 0.035 inch

<u>Oxidizer Tube (90 Tubes)</u>
Tube OD = 0.055 inch
Tube ID = 0.025 inch
Tube wall thickness = 0.015 inch

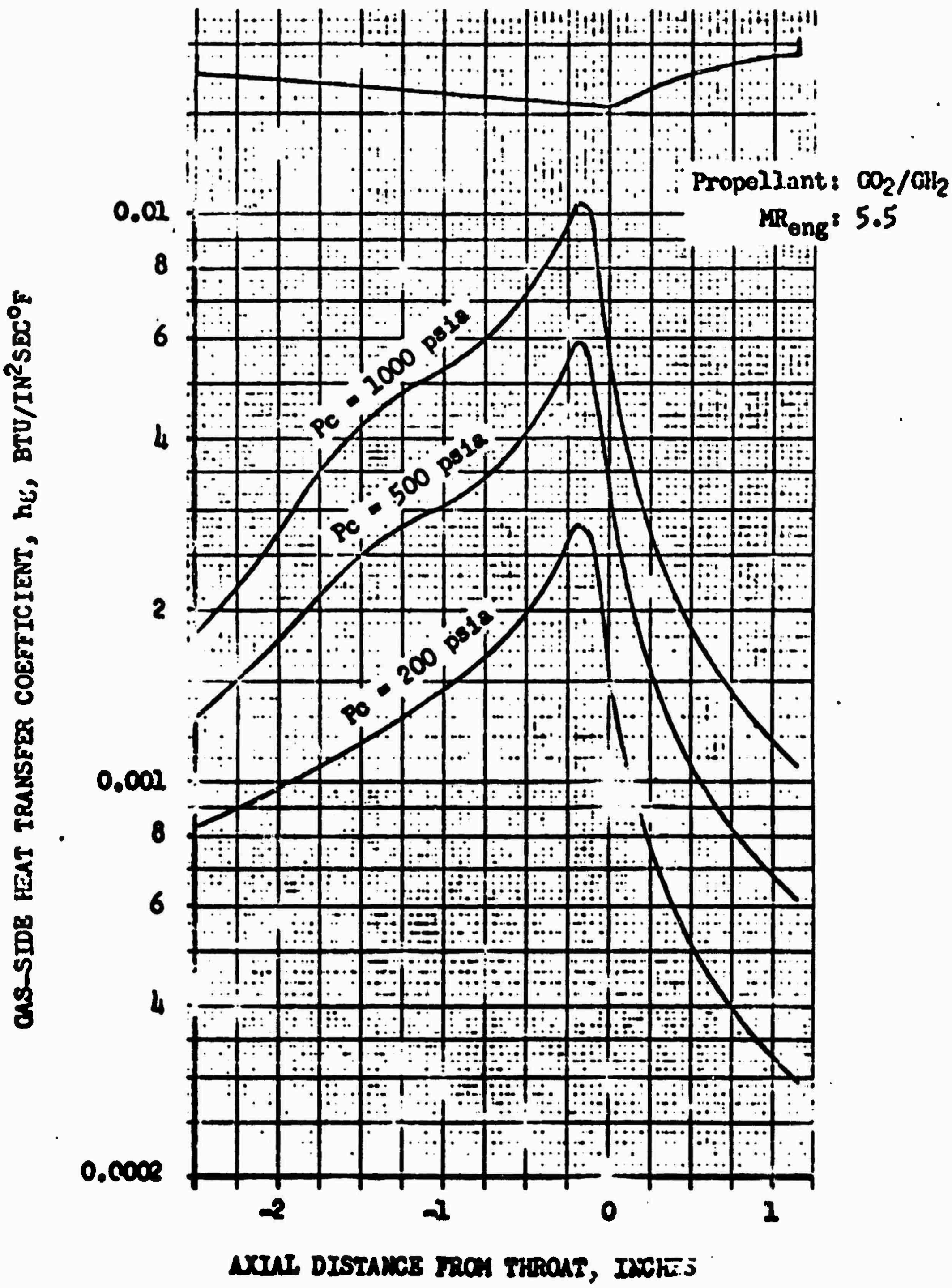


Figure 60. Double-Panel Combustor Gas-Side Heat Transfer Coefficient Distribution

The combustor inner and outer body coolant passage dimensions are noted in Fig. 61. The combustor side-panel coolant passage dimensions are shown in Fig. 62. The maximum combustor wall temperature occurs on the inner body at the peak heat flux location (X = -0.2 inch). At this location, the oxygen absorbs approximately 20 percent of the heat input and has a heat flux of 7 Btu/in.^{2}-sec. The axial variation of maximum gas-side wall temperature on the inner body is shown in Fig. 63. As shown in Fig. 64, a unique feature of the selected circuit configuration is that the maximum combustor wall temperature (located on the inner body at X = -0.2 inch) decreases as the thrust chamber is throttled.

Combustor, Single Panel. The combustor is assembled from a basic, single-piece NARLoy investment casting to which NARLoy closure sheets are brazed to form the complete rectangular coolant passages as illustrated in Fig. 65.

The selection of a NARLoy investment casting, rather than a conventionally machined combustor, was based on a lower cost for the casting as compared to the machined detail part. The NARLoy material was selected because of good castability, brazeability, high thermal conductivity, required material properties at elevated temperatures, and extensive experience with this material.

A tubular wall combustor, containing NARLoy tubes, was evaluated, but was rejected due to complexity of manifolding and manufacturing cost.

The major requirement for design of the combustor is the definition of the heat transfer conditions that exist in the combustor during operation and the regenerative cooling arrangement required to provide the combustor life

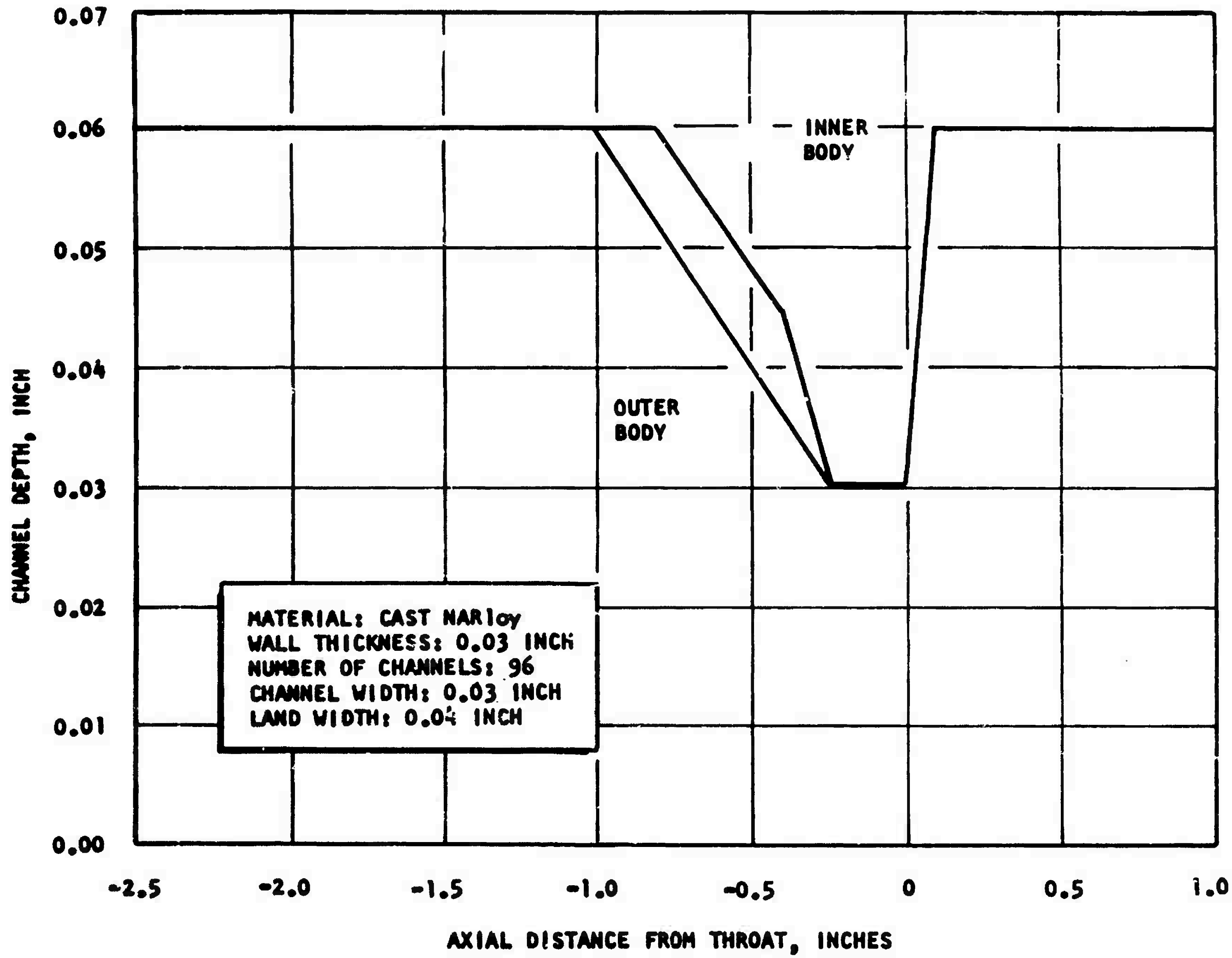


Figure 61. Double-Panel Outer and Inner Body Channel Dimensions

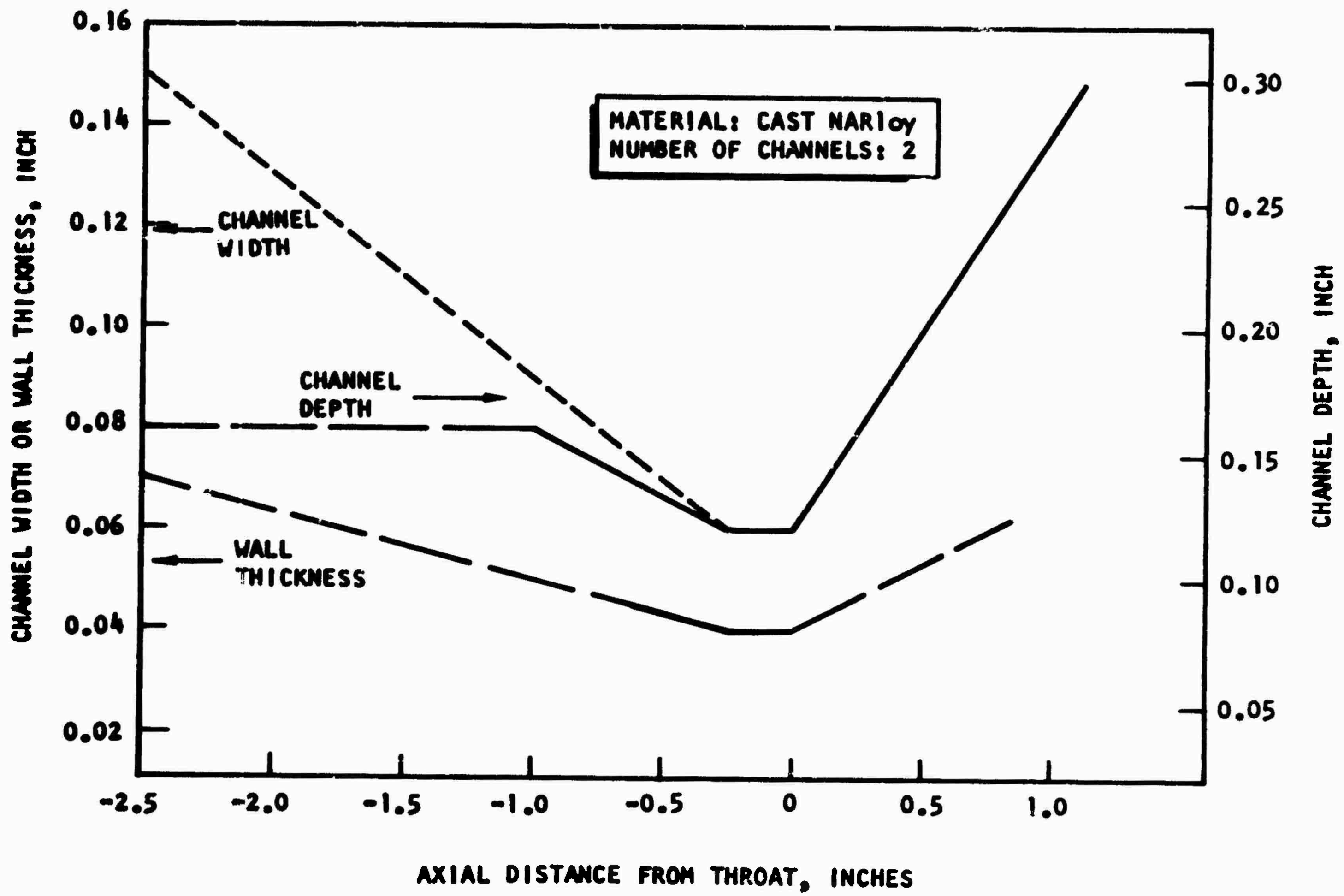


Figure 62. Double-Panel Side-Panel Channel Dimensions

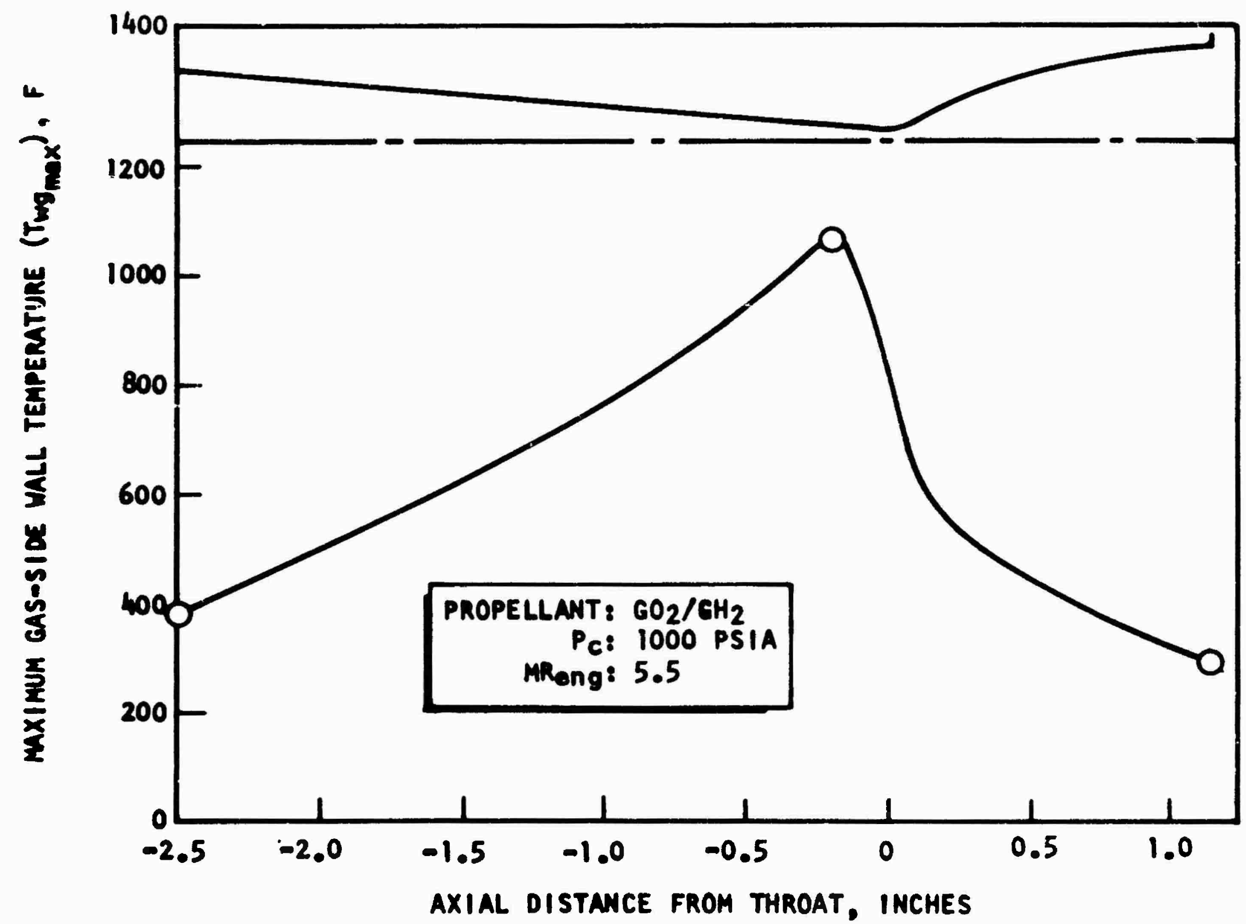


Figure 63. Double-Panel Combustor Inner Body Wall Temperature Distribution

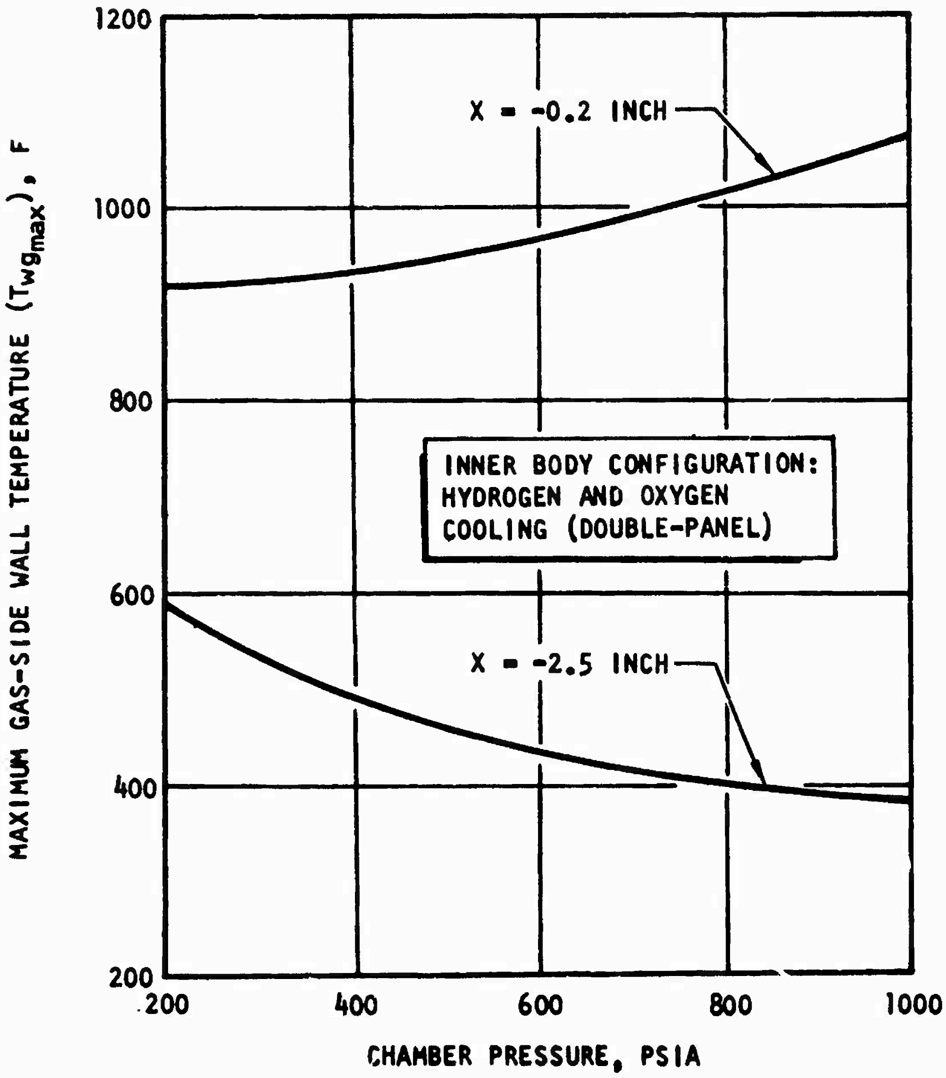


Figure 64. Inner Body Throat and Injector Region Wall Temperature Variation With Chamber Pressure

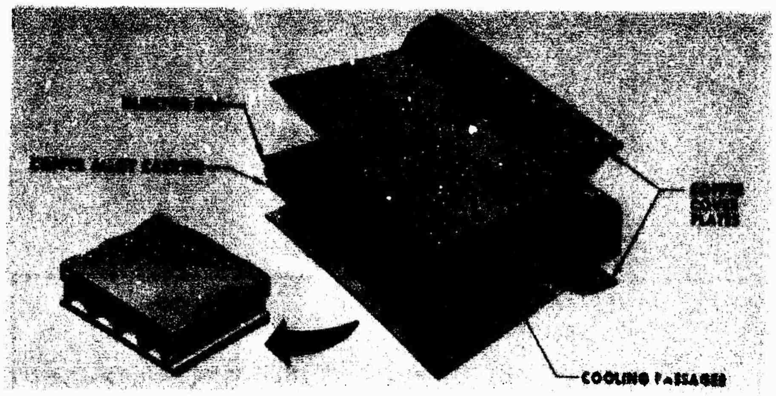


Figure 65. Single-Panel Thrust Chamber Segment

requirements. There are several different combustor-cooling circuits which are being evaluated for the single-panel thrust chamber design. The coolant circuit noted in Fig. 66, a single-pass, nozzle first-combustor last type, is currently being considered for the single-panel combustor.

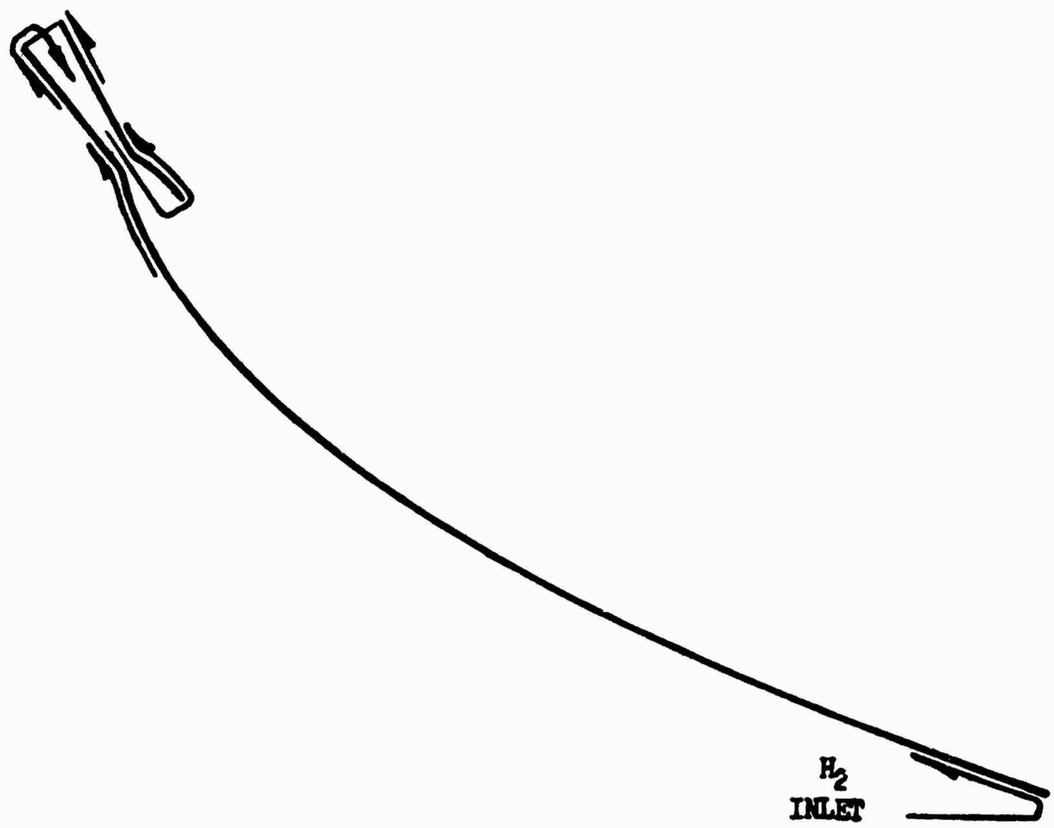


Figure 66. Single-Panel Demonstrator Thrust Chamber Cooling Circuit

With this circuit, the hydrogen enters the tubular nozzle cooling passages at the nozzle exit. After single up-pass cooling of the nozzle and thrust chamber inner body, the combustor side panels are down-pass cooled and, finally, the combustor outer body is up-pass cooled to complete the circuit. Down-pass signifies an injector-to-throat direction, and up-pass is the reverse. One undesirable feature of this circuit is that the coolant bulk temperature is near maximum at the outer body upper combustor location, and high combustor hot-gas side wall temperatures will result if high local heat transfer rates exist. A combustor first/nozzle last cooling circuit also was thoroughly evaluated because the coolant in the combustor region has not absorbed the nozzle heat yet but, due to reduced coolant temperatures at the outer body throat (80 to 150 R) (which, with hydrogen, is accompanied by a reduced cooling efficiency) very high combustor gas-side wall temperatures would be encountered over the throttle range at the outer body throat.

Following selection of the coolant circuit, sizing of the coolant passages was completed. The flow area at each selected axial station must be analytically determined so that coolant pressure loss is minimized and the combustor hot-gas temperature is low enough to satisfy the life requirements. The determination of the coolant passage size is accomplished by use of either an analytically predicted hot-gas side wall heat transfer film coefficient or an experimental, hot-fire test with calorimetry chamber, determined film coefficient. The combustor coolant passage dimensions were designed using the heat transfer coefficient distributions in Fig. 67 and 68. The channel dimensions and wall thicknesses are shown in Fig. 69 through 71. Channel dimensions at the minimum flow area are:

Inner Body (72 Channels)

Channel Width = 0.040 inch

Channel Depth = 0.025 inch

Land Width = 0.040 inch

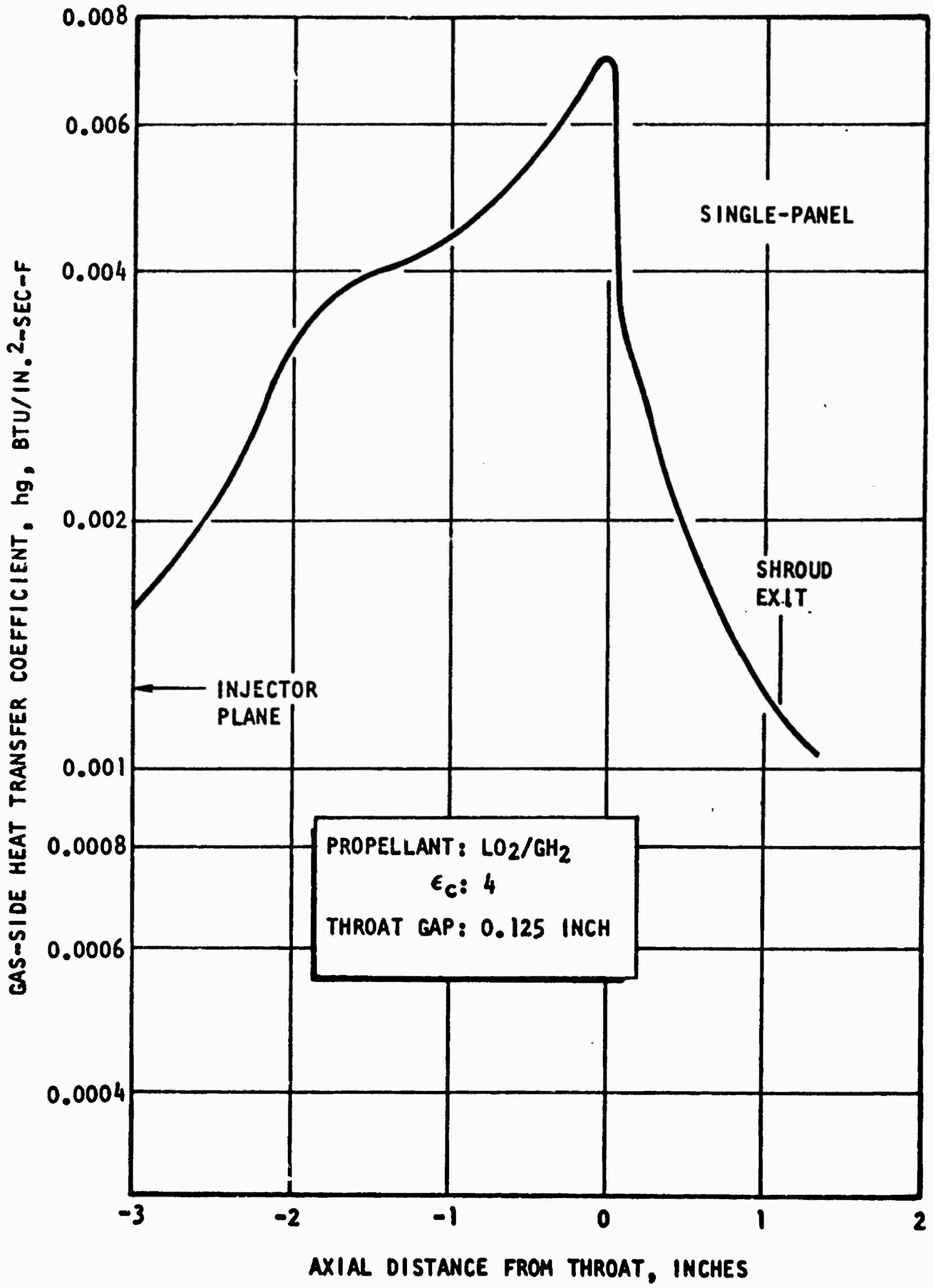


Figure 67. Analytically Predicted Combustor Gas-Side Heat Transfer Coefficient Distribution (P_c = 750 psia, MR = 5.5)

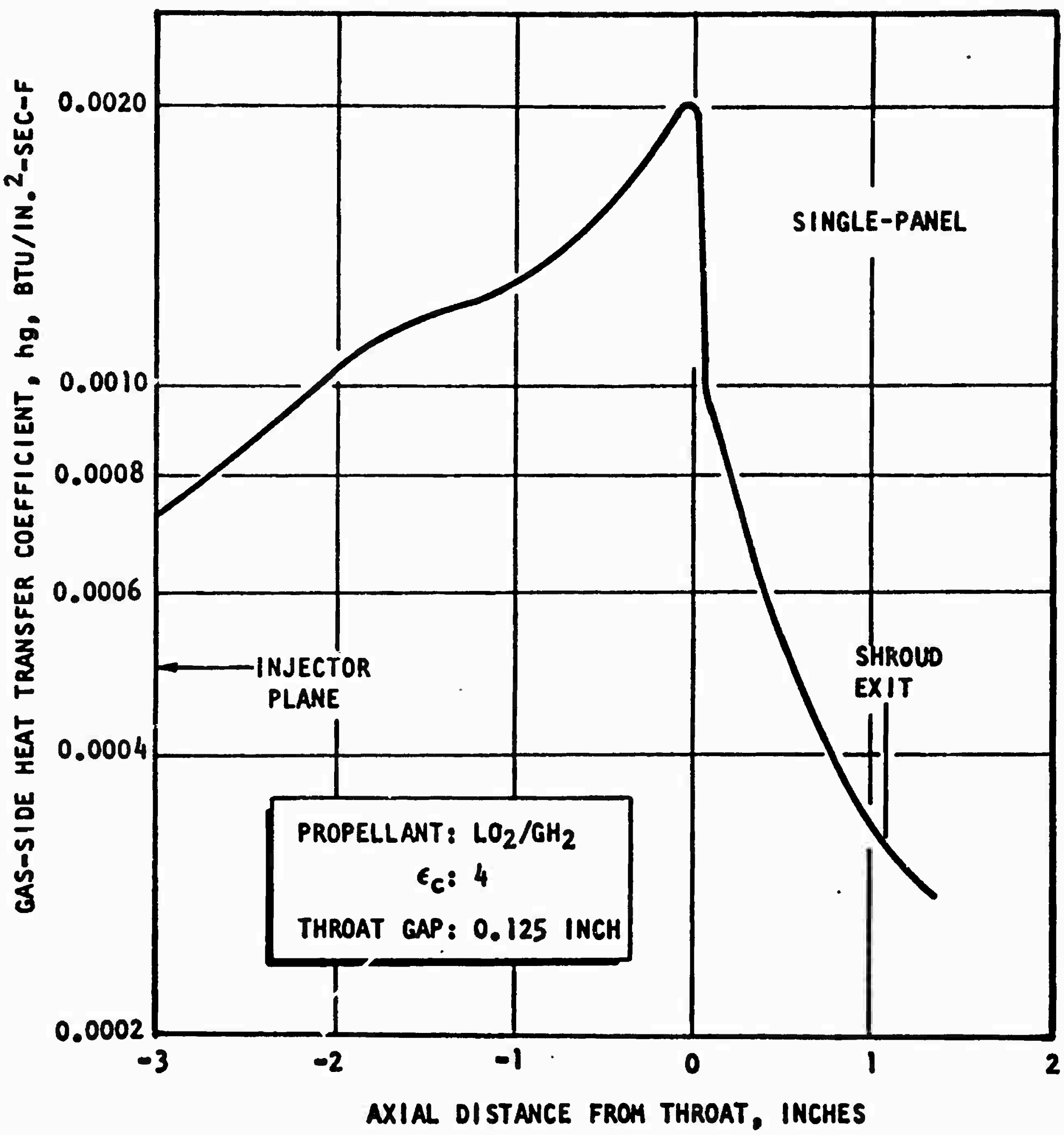


Figure 68. Analytically Predicted Combustor Gas-Side Heat Transfer Coefficient Distribution (P_c = 150 psia MR_{eng} = 5.5)

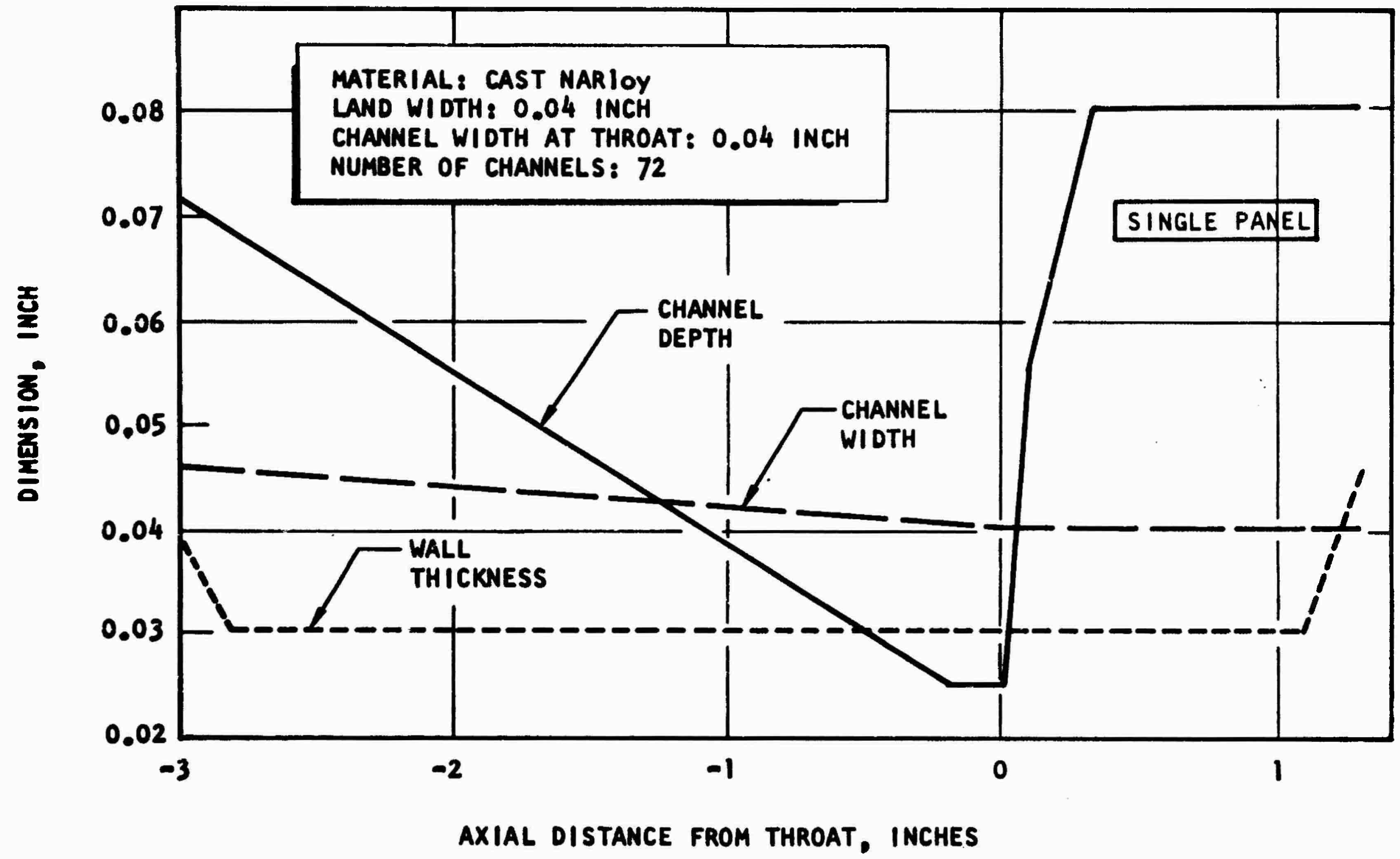


Figure 69. Combustor Inner Body Channel Dimensions

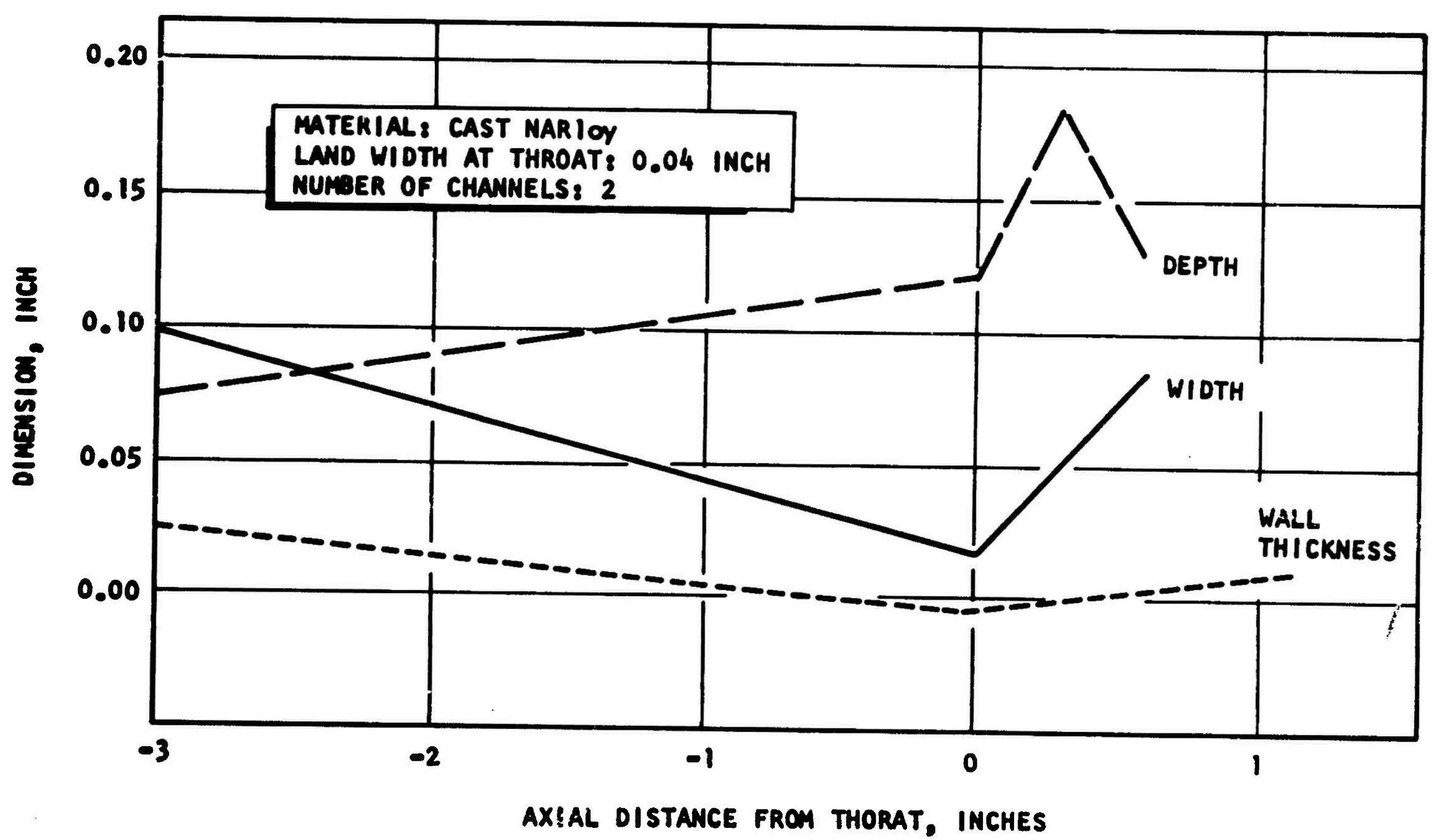


Figure 70. Side Panel Channel Dimensions

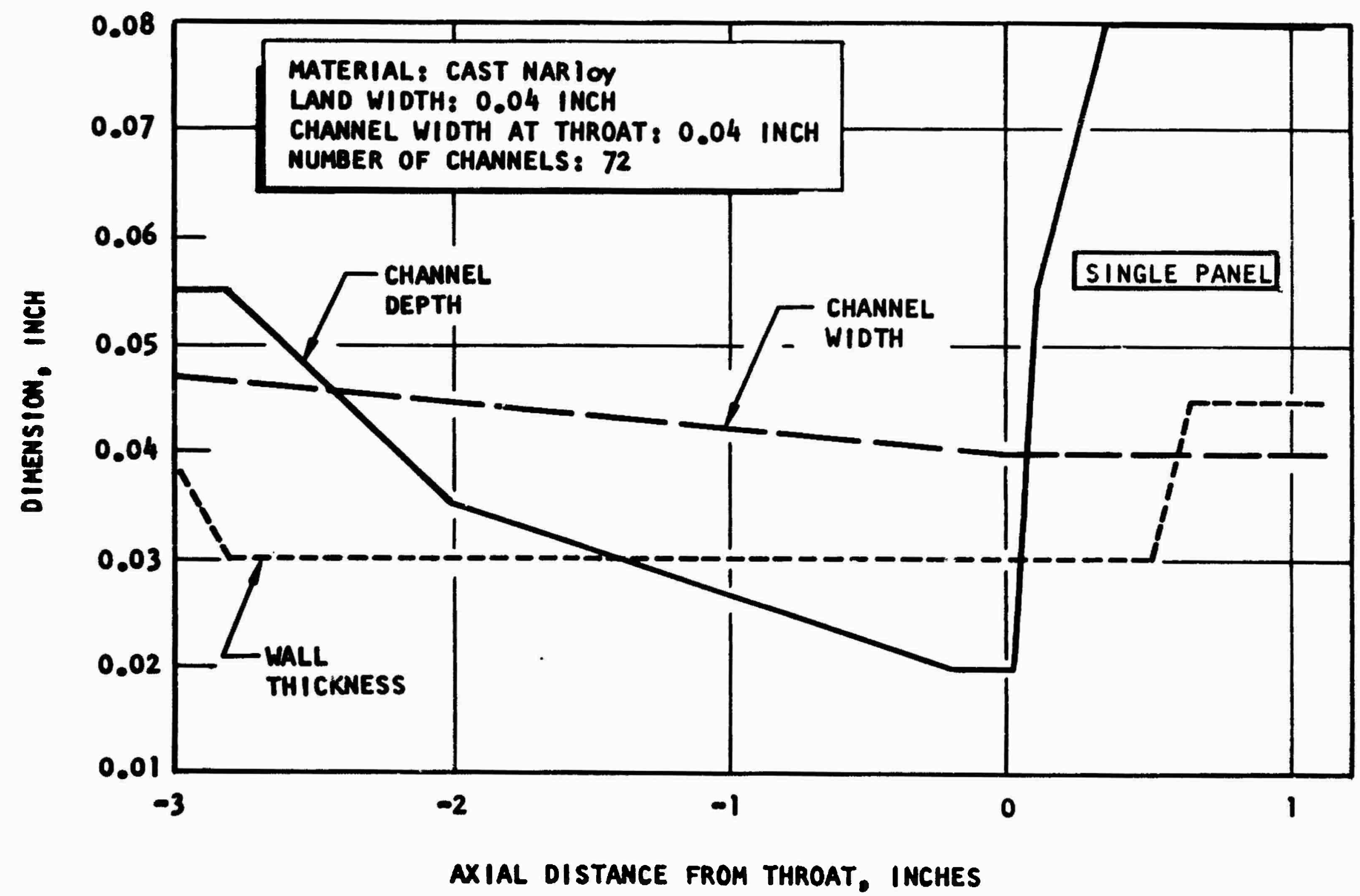


Figure 71. Combustor Outer Body Channel Dimensions

Side Panel (2 Channels)

Channel Width = 0.067 inch

Channel Depth = 0.172 inch

Land Width = 0.040 inch

Outer Body (72 Channels)

Channel Width = 0.040 inch

Channel Depth = 0.020 inch

Land Width = 0.040 inch

The heat transfer analysis was conducted for the selected coolant passage designs at 750- and 150-psia chamber pressure. The coolant temperature rise, coolant pressure drop, and minimum two-dimensional gas-side wall predicted temperature were computed. The axial distribution of the maximum two-dimensional hot-gas side wall temperatures also were computed for the combustor inner and outer bodies (Fig. 72 and 73).

The combustor geometry also was evaluated during the experimental program. An injector width of 0.5 and 0.6 inch and combustion chamber lengths of 3.0 and 4.0 inches were evaluated for their effect on heat transfer and performance. The 0.5-inch-wide injector and 3.0-inch combustion chamber length were selected. No benefit was apparent with the 0.6-inch-width injector face, and the 4.0-inch-long combustor resulted in excessive heat rejection rates due to increased area.

Injector, Double-Panel. The injector assembly is welded to the thrust chamber to provide a lightweight assembly. The selection of the injector was based on an experimental test and development program. The injector design requirements were:

1. Operation over chamber pressure range of 200 to 1000 psia
2. η_{C*} of 97 percent minimum over the throttle range
3. Dynamically stable over complete operating range

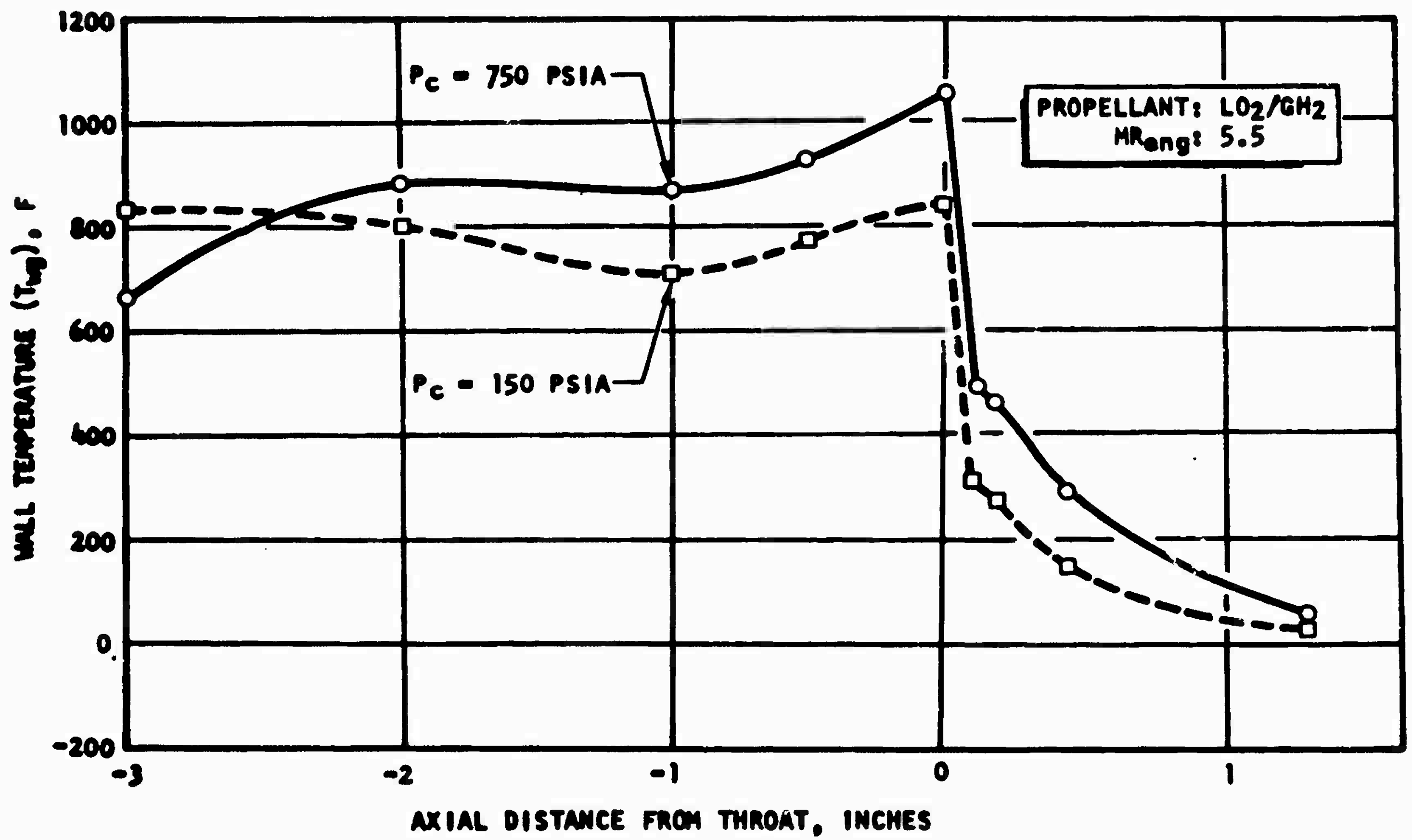


Figure 72. Single-Panel Combustor Inner Body Wall Temperature Distribution

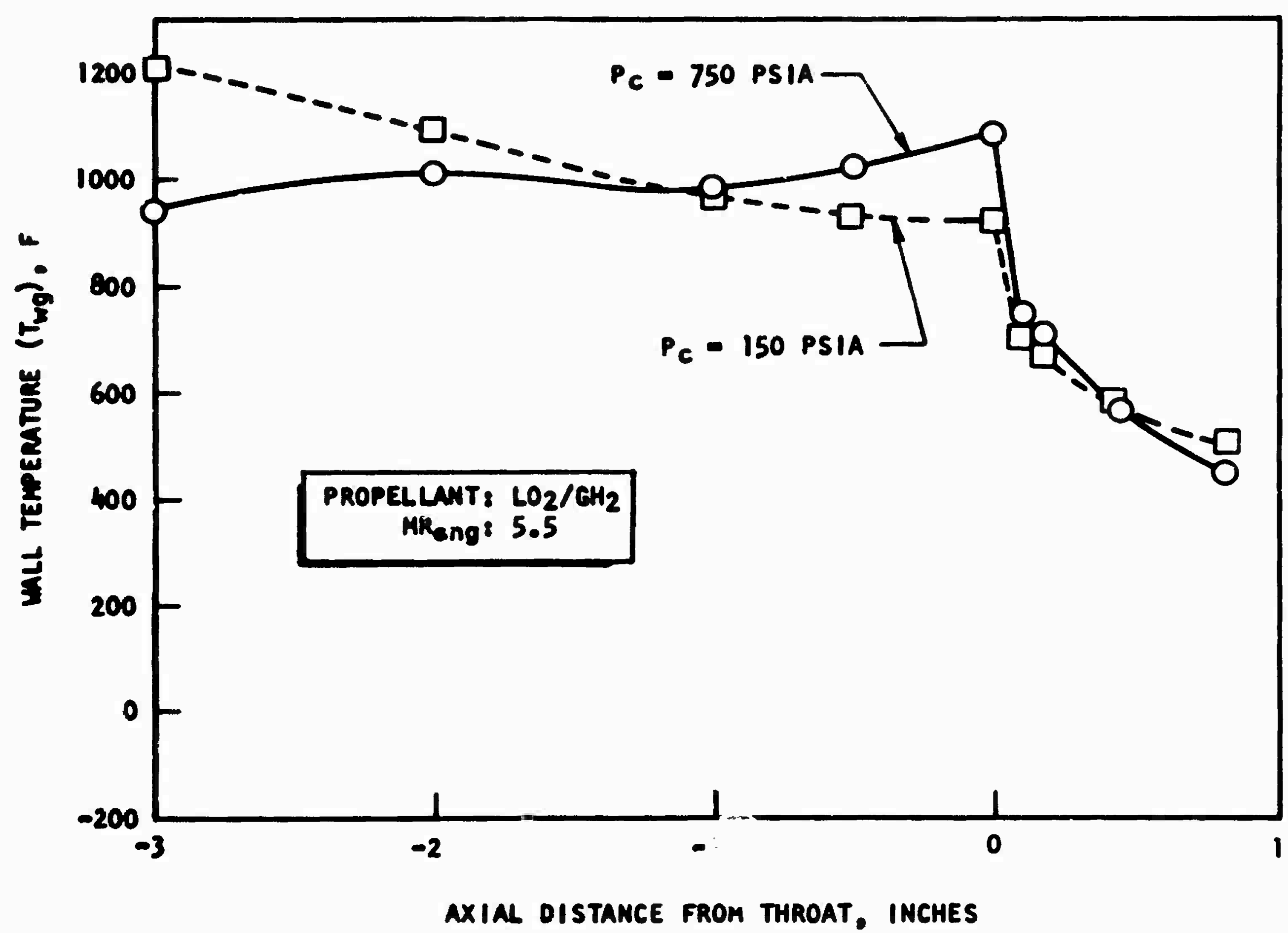


Figure 73. Single-Panel Combustor Outer Body Wall Temperature Distribution

4. Durability to satisfy engine life requirements

5. Smooth ignition and chamber pressure transient characteristics

6. No excessive injector-induced combustion chamber wall streaking and uniform heat transfer into the wall.

Three candidate injector patterns were extensively tested and developed (Fig. 74 through 76). Two injector types (triplet and trislot) demonstrated comparable performance and heat transfer characteristics during test. The triplet element type (Fig. 74) was selected because of lower fabrication costs. The design characteristics are as follows:

1. 50 elements, 2 rows

2. 0.5-inch injector face width

3. Hydrogen injection velocity of 1400 ft/sec at the 1000-psia chamber pressure design point

4. Oxygen injection velocity of 500 ft/sec at the 1000-psia chamber pressure design point

5. A hydrogen-oxygen-hydrogen triplet pattern with the resultant fans oriented parallel to the hot-gas walls, inner and outer

6. An included impingement angle between hydrogen streams of 75 degrees

7. The impingement point is located 0.090-inch downstream of the injector face

8. Equal element-to-element spacing circumferentially

9. Minimum possible element-to-wall gap

10. Injector face plate of NARloy or a material with equivalent or higher thermal conductivity and body of 304L CRES

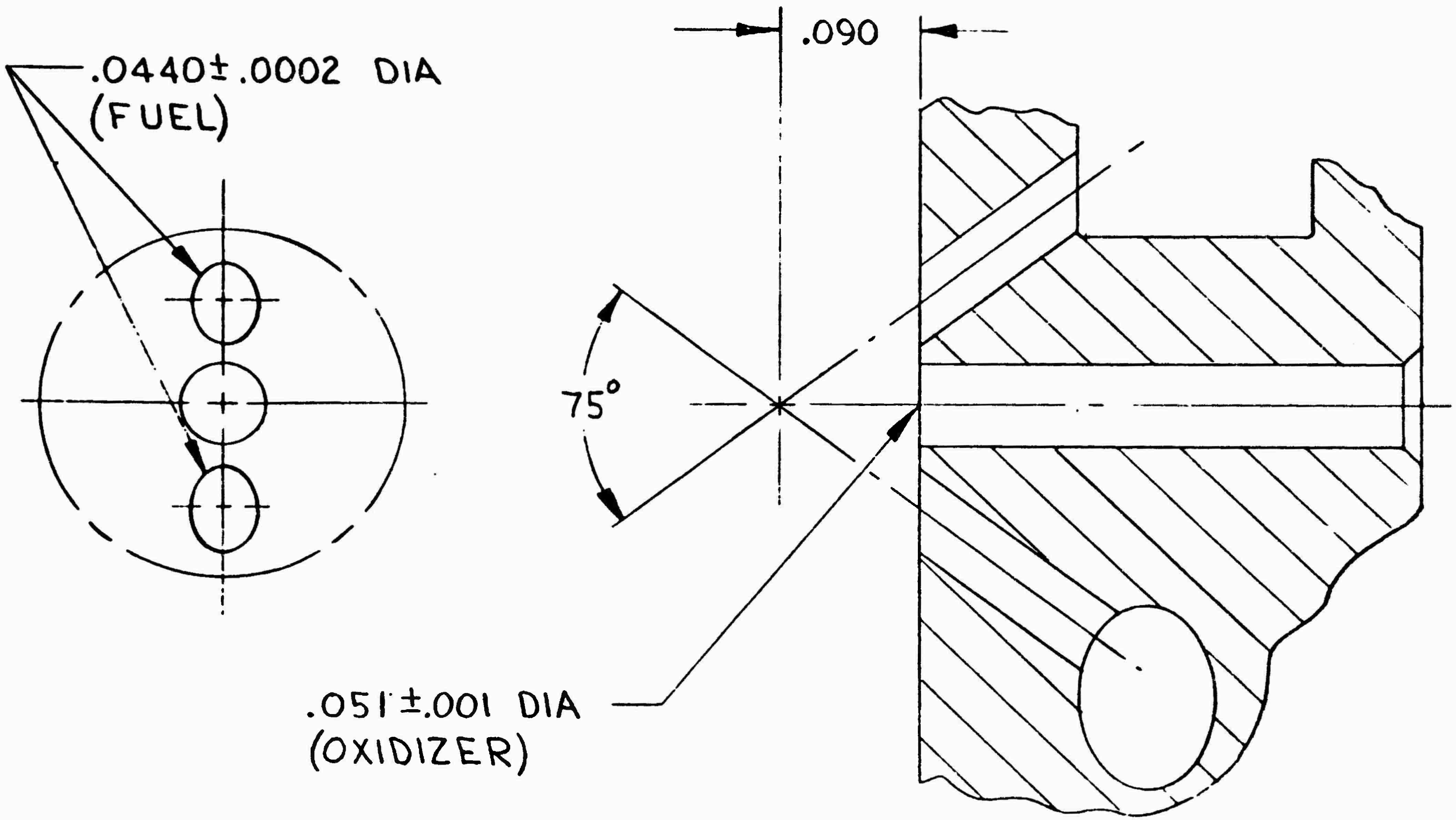


Figure 74. Triplet Injector Element (3 Orifices/Element, 50 Elements/Segment, 1224 Elements/Thrust Chamber)

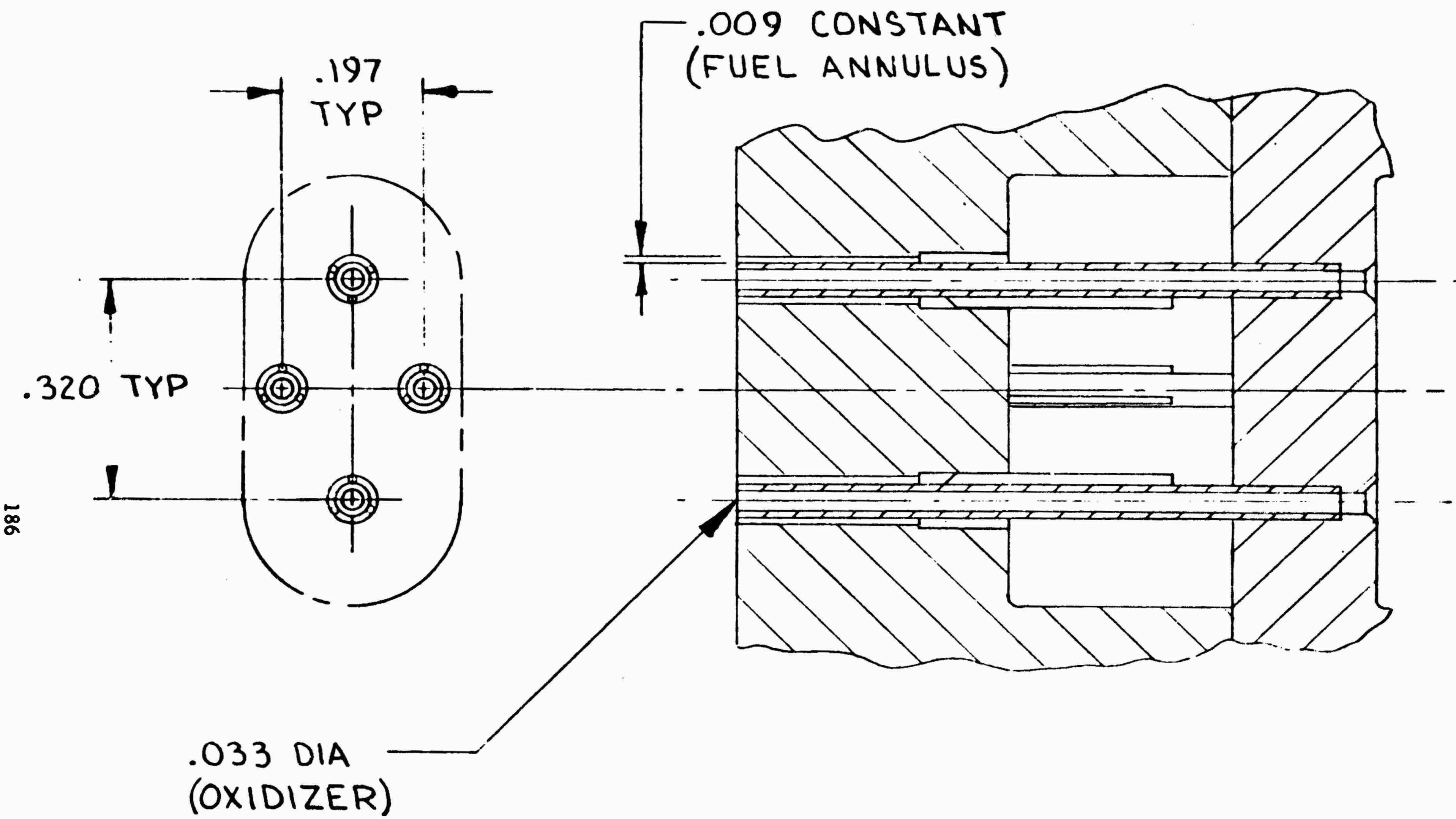


Figure 75. Concentric Injector Elements (101 Elements/
Segment, 2424 Elements/Thrust Chamber

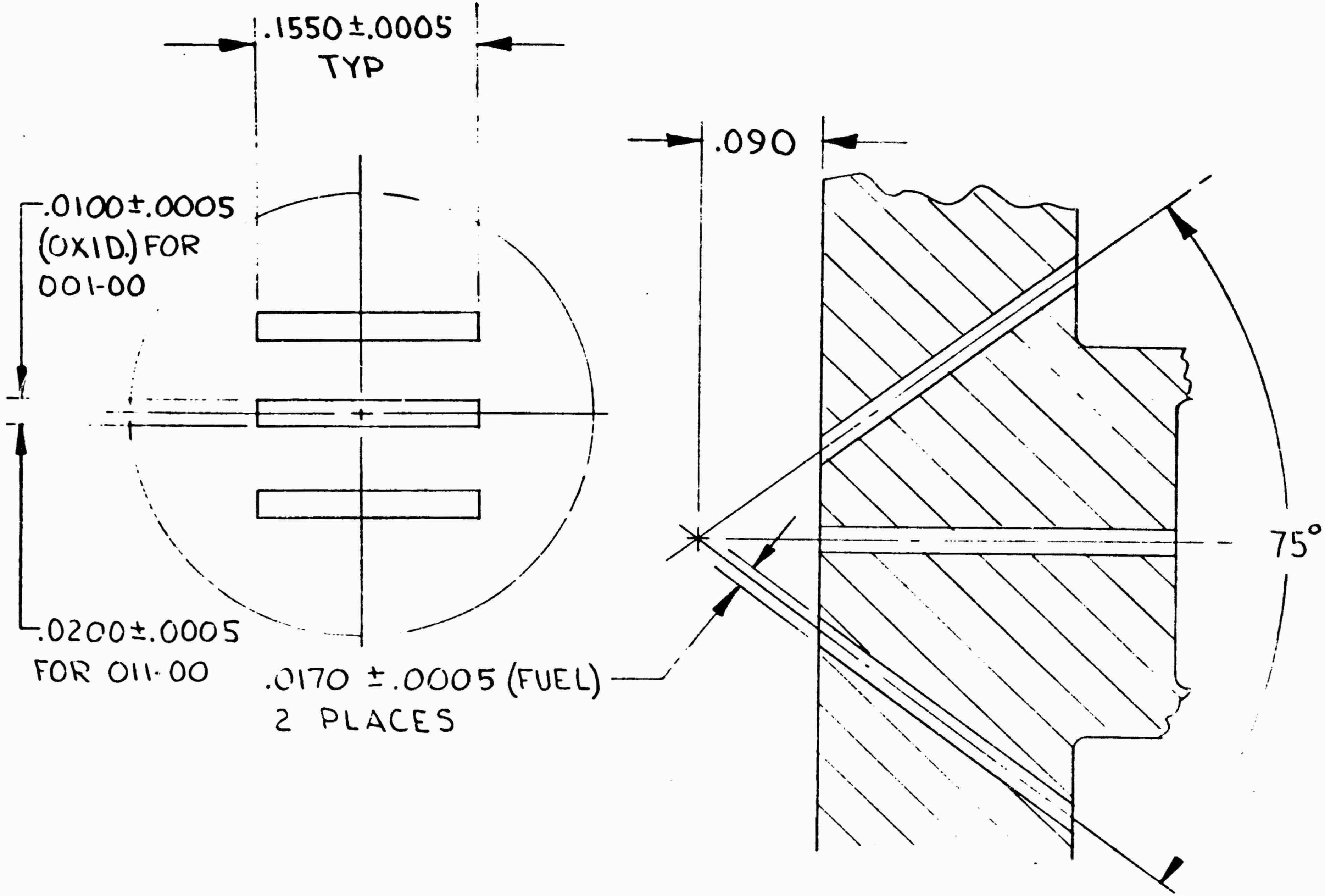


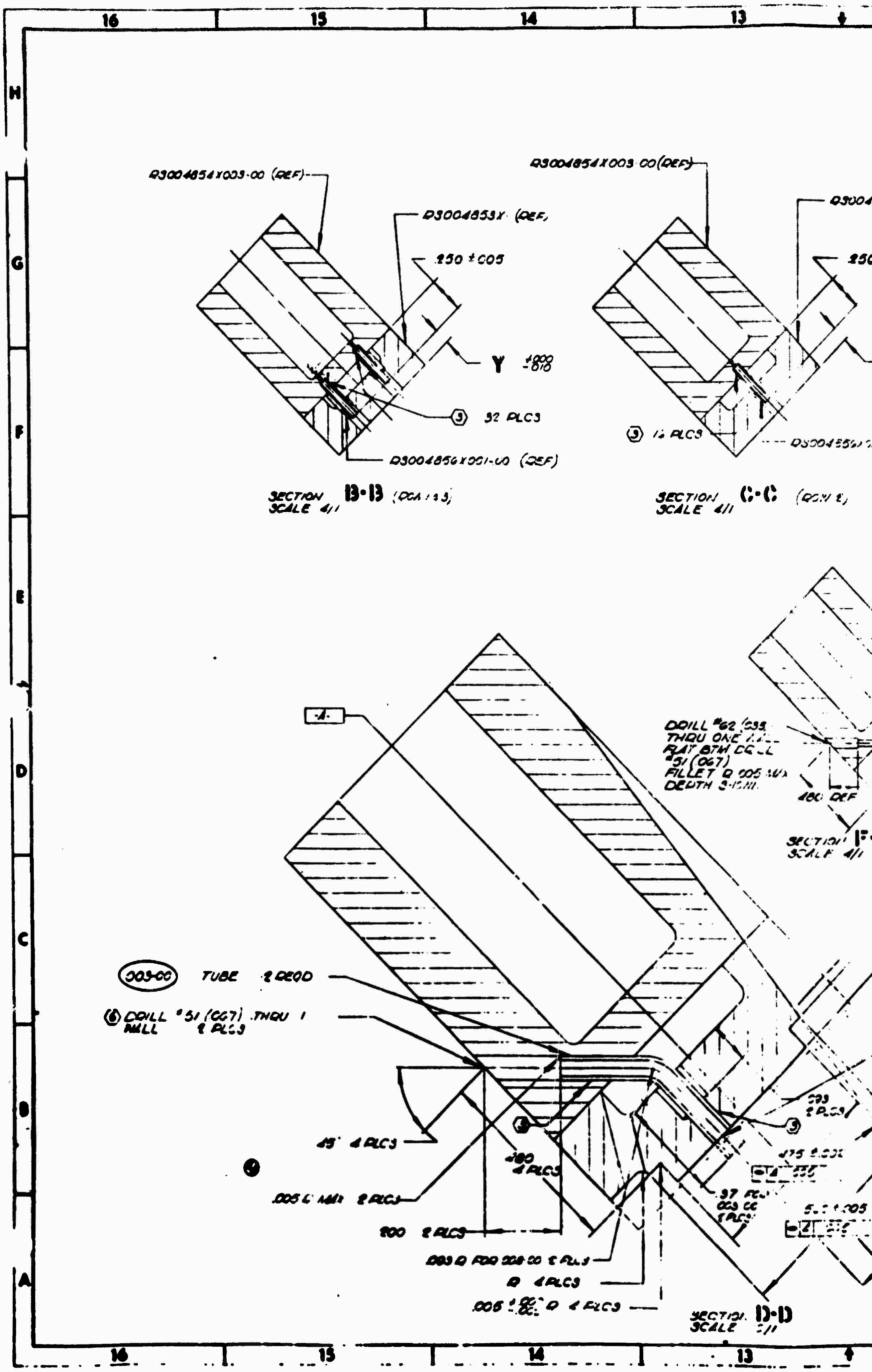
Figure 76. Trislot Injector Element (3 Orifices/Element, 30 Elements/Segment, 720 Elements/Thrust Chamber)

Injector, Single-Panel. The injector assembly is welded to the thrust chamber to provide a lightweight assembly. The selection of the injector was based on an experimental test and development program. The injector design requirements are:

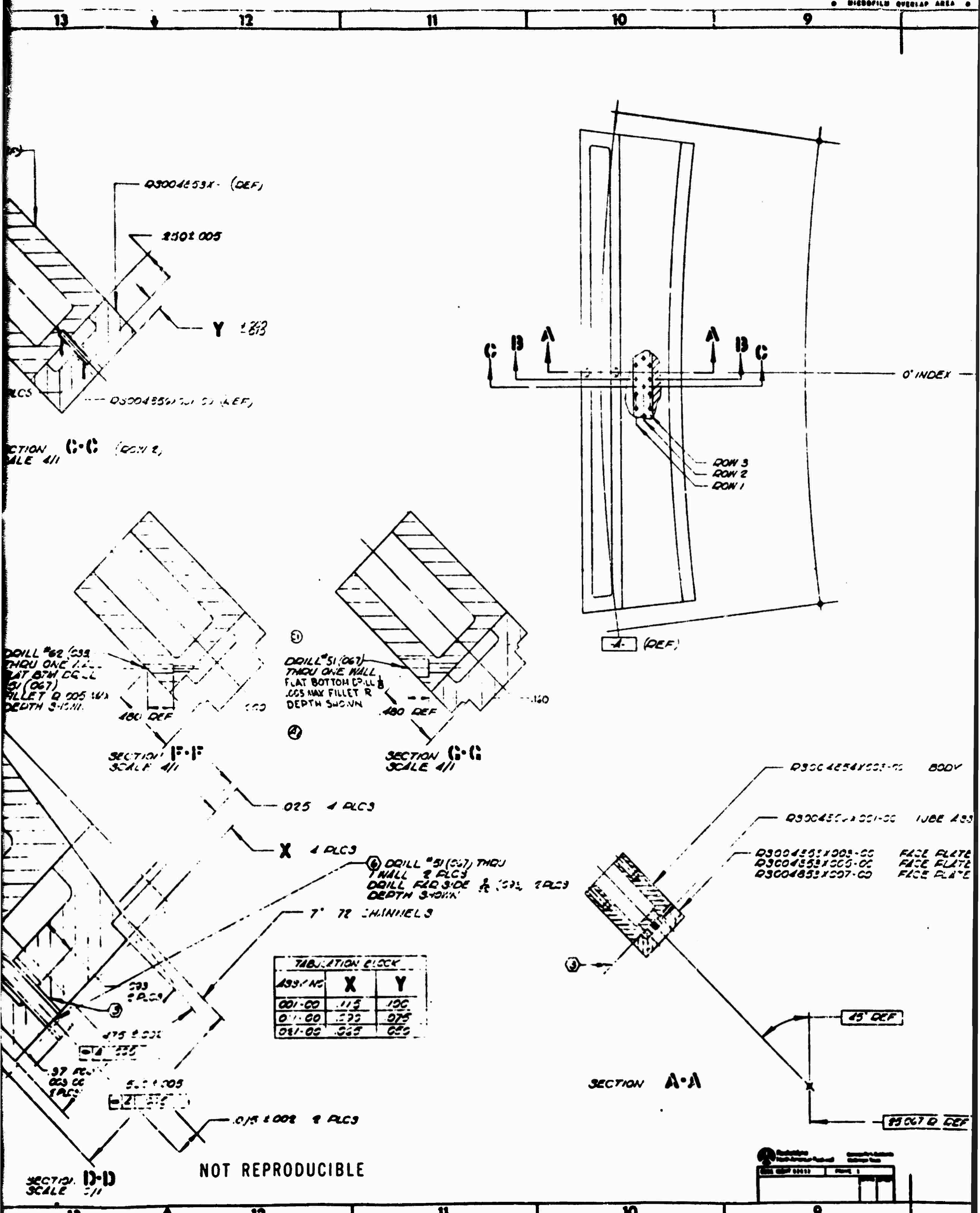
1. Operates over chamber pressure range of 150 to 750 psia
2. Provides a η_{c*} of 97 percent minimum over the throttle range
3. Dynamically stable over complete operating range
4. Durable and meets service life requirements
5. Smooth ignition and chamber pressure transient characteristics
6. No excessive injector induced combustion chamber wall streaking and uniform heat transfer into the wall

Three candidate injector paterns were extensively tested and developed. From this program, a concentric orifice element type injector was selected. The design characteristics are shown in Fig. 77 and noted in the following:

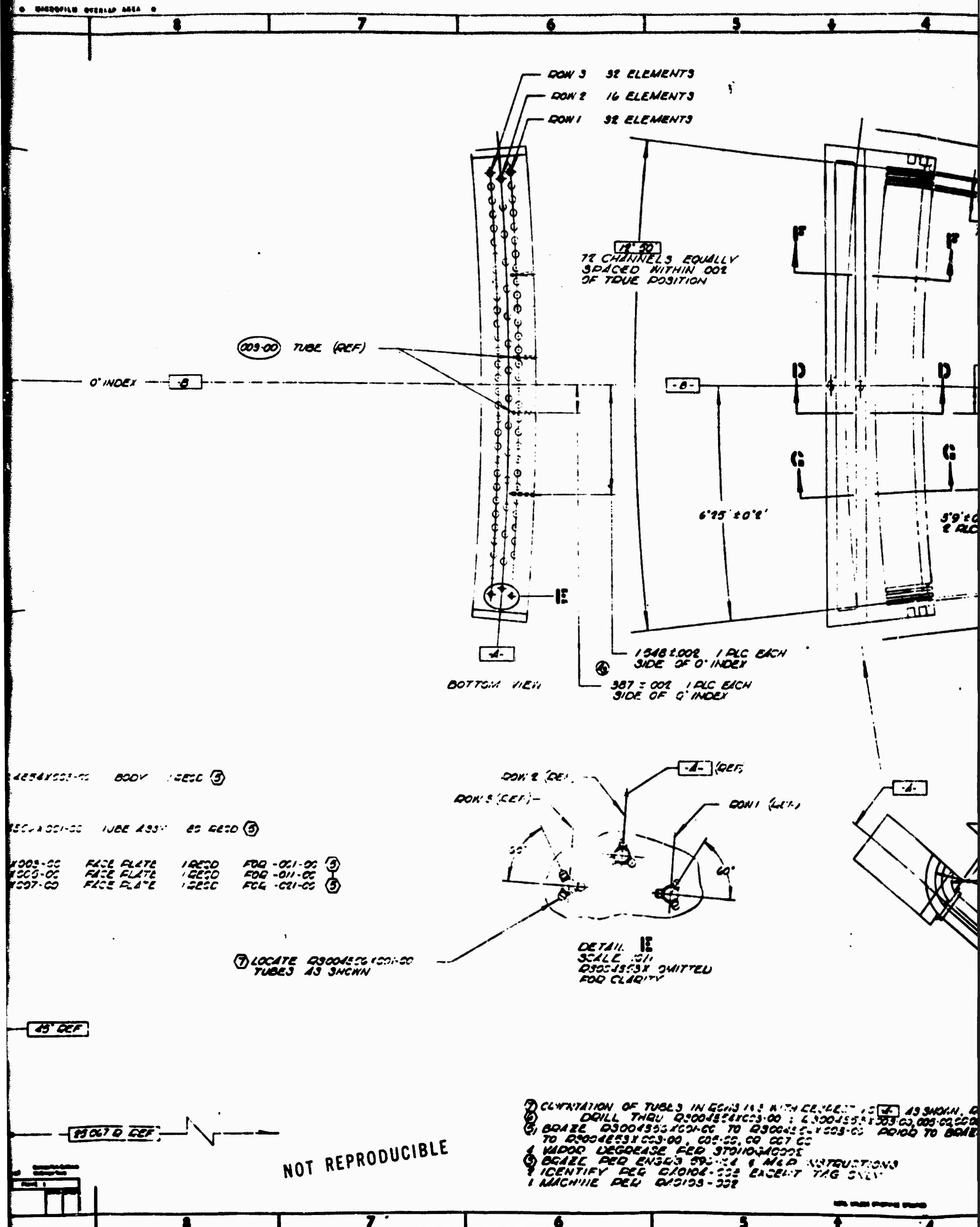
1. 80 elements, 3 rows
2. Oxidizer post wall thickness of 0.005 inch
3. Constant thickness oxidizer post wall; no exit flare or divergence
4. 0.5-inch injector width
5. Hydrogen annulus velocity at the predicted injection temperature of 1250 ft/sec at 750-psia chamber pressure
6. Oxidizer post discharge velocity of 50 ft/sec at 750-psia chamber pressure
7. An oxidizer post recess of 0.100 inch to obtain high performance
8. A wall-to-element spacing of 0.090 inch
9. Injector face plate of NARLoy for optimum face cooling and the body of Armco 21-6-9 for high strength and minimum weight



16
15
14
13
H
G
F
E
D
C
B
A
Q3004854X003-00 (REF)
Q3004853X (REF)
.250 ±.005
Y
32 PLCS
Q3004856X001-00 (REF)
SECTION B-B
SCALE 4/1
SECTION C-C
SCALE 4/1
Q3004854X003-00(REF)
-A-
003-00 TUBE 2 REQD
DRILL #51 (067) THRU 1 WALL 2 PLCS
45° 4 PLCS
.480 4 PLCS
.005 MAX 2 PLCS
.200 2 PLCS
4 PLCS
480 REF
SECTION D-D


13
12
11
10
9
ROW 3
ROW 2
ROW 1
0° INDEX
(REF)
SECTION C-C
SECTION F-F
SCALE 4/1
SECTION G-G
SCALE 4/1
DRILL #51 (.067)
THRU ONE WALL
FLAT BOTTOM DRILL
.005 MAX FILLET R
DEPTH SHOWN
.480 REF
.160
.025 4 PLCS
X 4 PLCS
7° 72 CHANNELS
BODY
SECTION A-A
45° REF
SECTION D-D
.015 ±.002 2 PLCS
TABULATION BLOCK
ASSY NO
X
Y
001-00
003-00
007-00
NOT REPRODUCIBLE

2



NOT REPRODUCIBLE

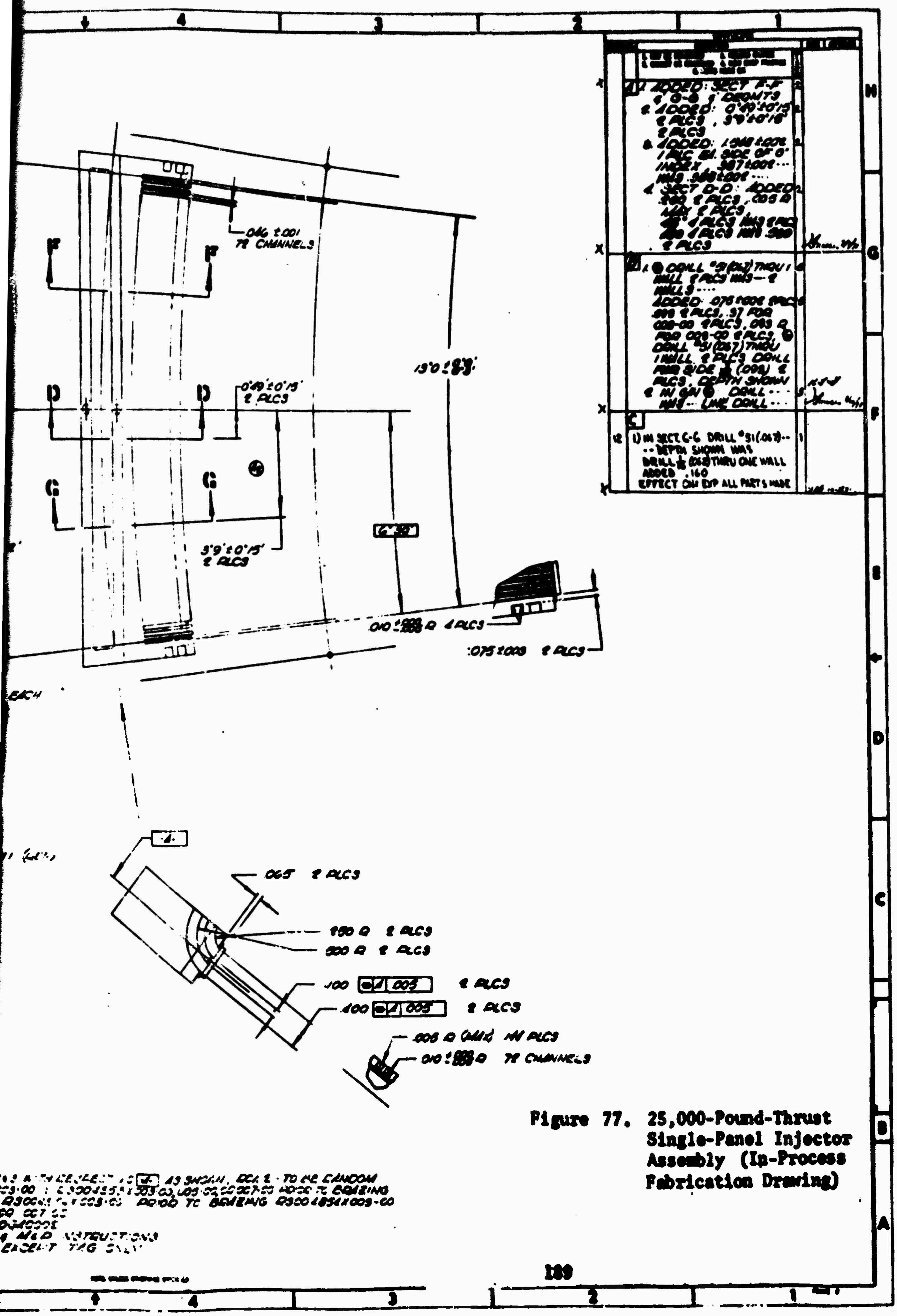


Figure 77. 25,000-Pound-Thrust Single-Panel Injector Assembly (In-Process Fabrication Drawing)

As noted previously, the injector configuration that was selected has been hot-fire tested and has demonstrated the noted characteristics.

Structural Support. A structural support system is required for both the single-panel and double-panel thrust chamber. The structural support system consists of two concentric rings, an inner and outer which restrain pressure forces within the combustor and transmit thrust forces generated by the combustor. The rings for the double-panel chamber assembly are more substantial because of the higher design point chamber pressure.

A cross section of the rings reveals four square corrugations with 1.125-inch thick caps. At 24 equally spaced positions on the rings, a pad provides for the through bolts that connect the inner and outer ring and constrain the combustors within the rings. These pads with ribs interrupt the corrugations, and with bolts installed, provide structural beam support and fixity between combustion segments.

The structural rings are machined from titanium alloy (Ti-6Al-2Sn-4Zr-2Mo) forgings. This alloy was chosen for its lightweight and high strength at room and elevated temperatures. The essentially open-face ribs of the corrugation allows access for machining and will be superior weightwise to honeycomb and other closed-type structures. A combination of conventional machining and EDM will be used to manufacture the structural support rings.

Nozzle Extension

A nozzle extension is required as an integral part of both the single-panel and double-panel thrust chamber assemblies. The nozzle extension is required to develop thrust due to pressure of the expanding gases acting on the nozzle surface, analogous to the divergent portion of a DeLaval nozzle. The part is regeneratively cooled in preference to ablative or radiative cooling because regenerative cooling results in the lightest weight structure.

Double-Panel Nozzle Extension. The extension is single down-pass regeneratively cooled by hydrogen introduced at the upper end nozzle-to-combustor transition ring. The extension is constructed of 1450 Haynes 188 tubes, which have approximately 0.120-inch OD and 0.005-inch wall thickness. The use of Haynes 188 material is necessary because of increased pressures and temperatures associated with the double-panel cooling circuit.

The analysis to define the coolant flow area at each axial station utilized the analytically predicted heat transfer film coefficients shown in Fig. 78. The analytically predicted gas-side wall temperature profile is shown in Fig. 79 . The nozzle tube dimensions are shown in Fig. 80.

Fabrication techniques are similar to those for the single-panel nozzle extension.

Single-Panel Nozzle Extension. The extension is single up-pass regeneratively cooled by hydrogen coolant introduced at the exit end. The nozzle extension provides an attachment place for the base closure assembly in addition to being weld-attached to the 24 thrust chambers. The extension is constructed of 1600 tubes approximately 0.060-inch OD and 0.005-inch wall thickness, of AISI 347 corrosion-resistant steel. The material has more than sufficient strength at the operating temperatures.

The analysis to define the coolant flow area at each axial station utilized the analytically predicted heat transfer film coefficients shown in Fig. 81. The analytically predicted gas-side wall temperature profile is shown in Fig. 82. Nozzle tube dimensions are shown in Fig. 83.

The nozzle extension is a furnace-brazed assembly. The coolant tubes are tapered, assembled on a mandrel, alloyed, and then furnace-brazed together. The coolant inlet manifold located at the nozzle exit end forms one of the hoop structural members and provides the attachment location for the base closure.

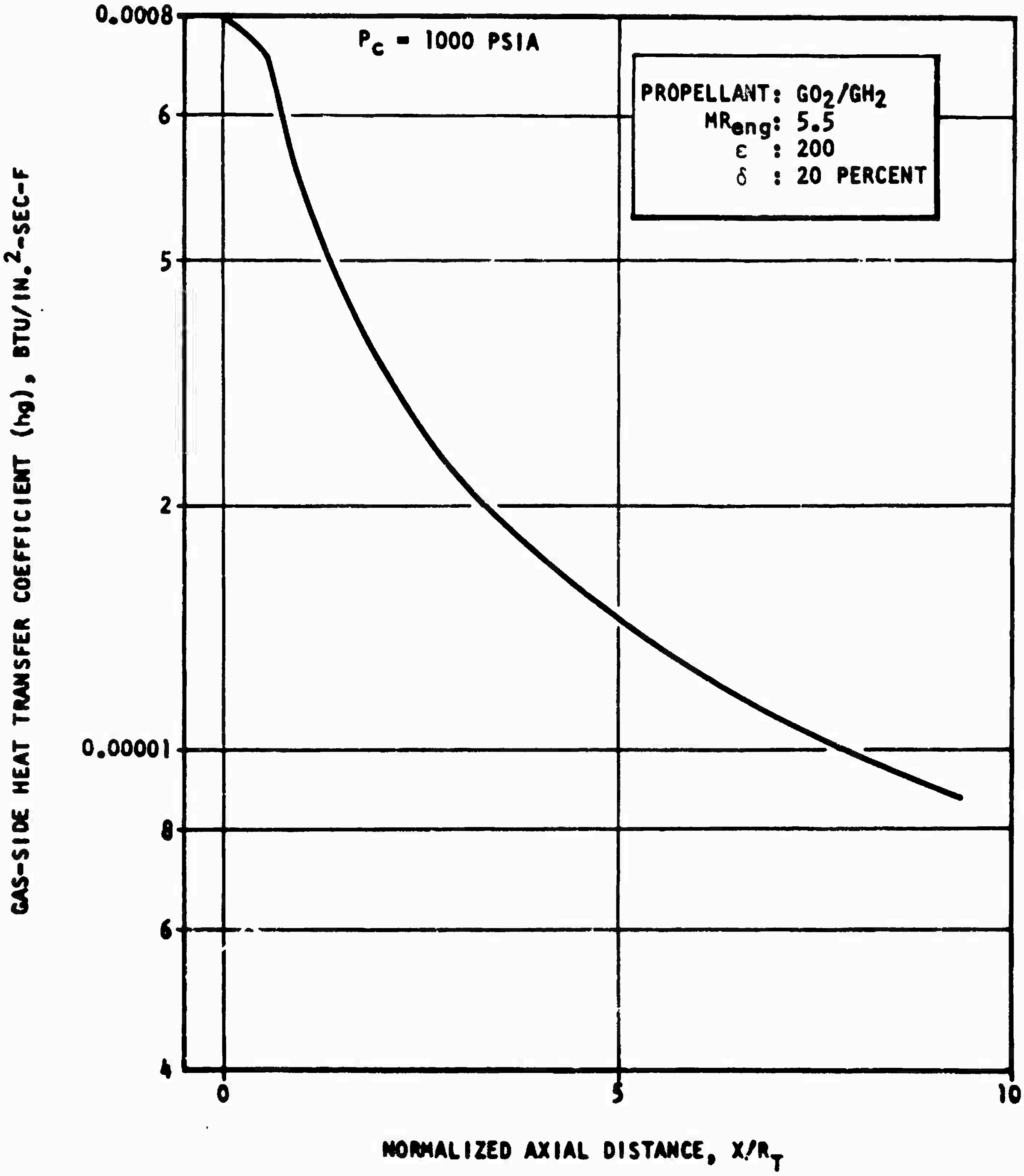


Figure 78. Double-Panel Nozzle Gas-Side Heat Transfer Coefficient Distribution

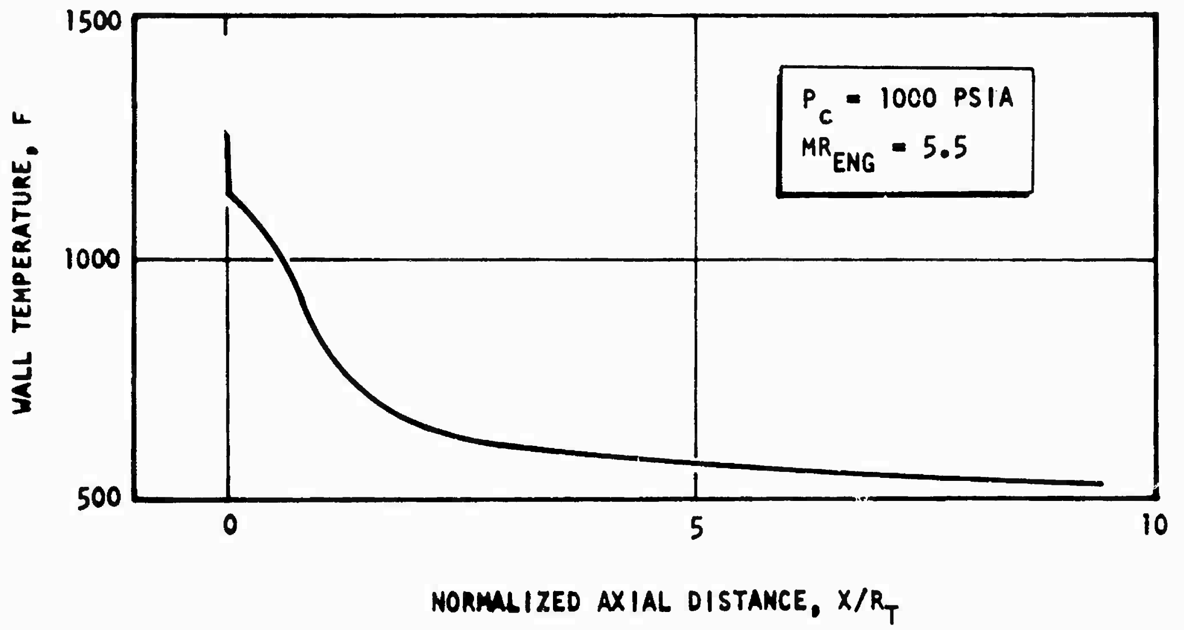


Figure 79. Double-Panel Nozzle Gas- Side Wall Temperature Distribution

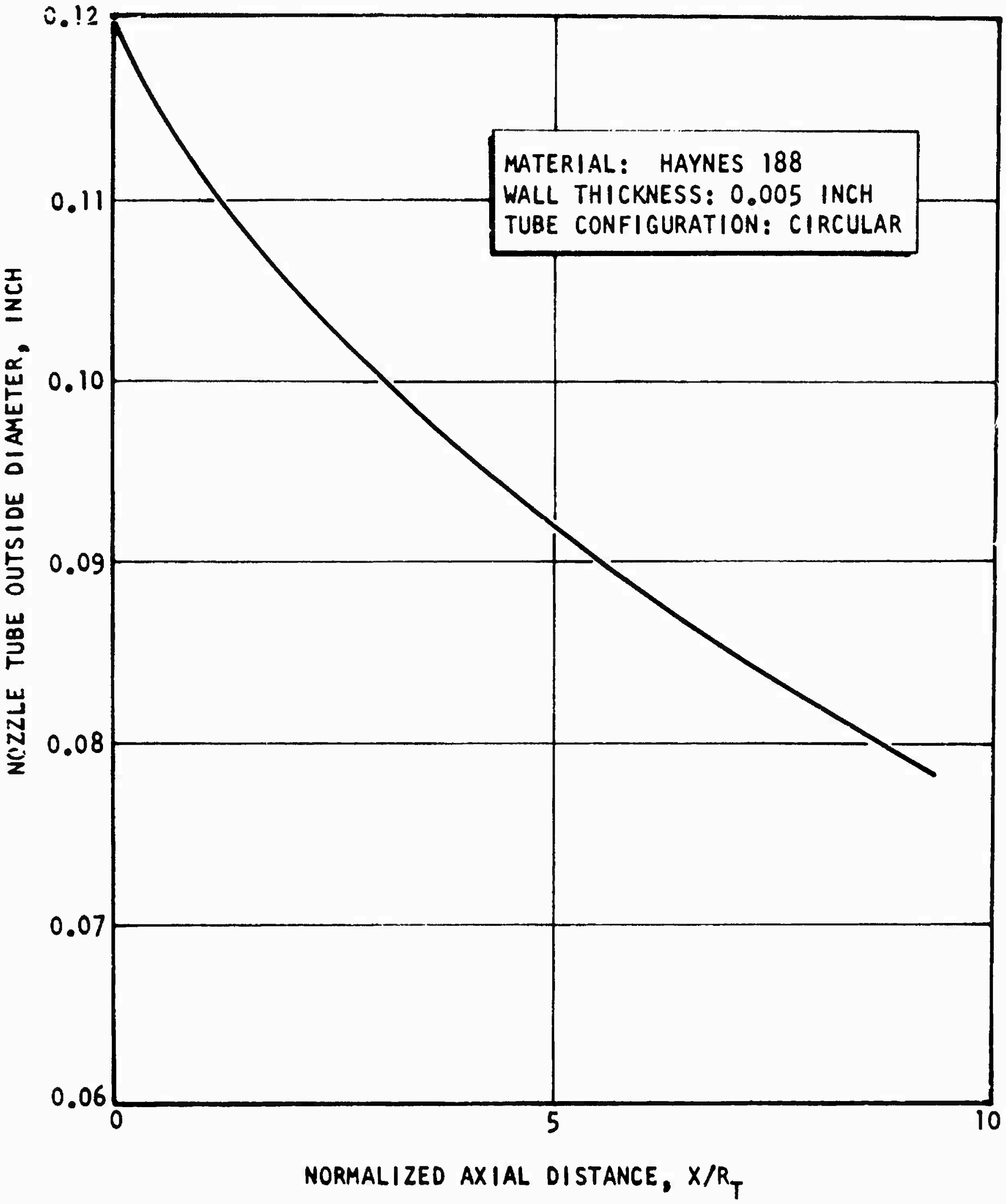


Figure 80. Double-Panel Nozzle Tube Dimensions

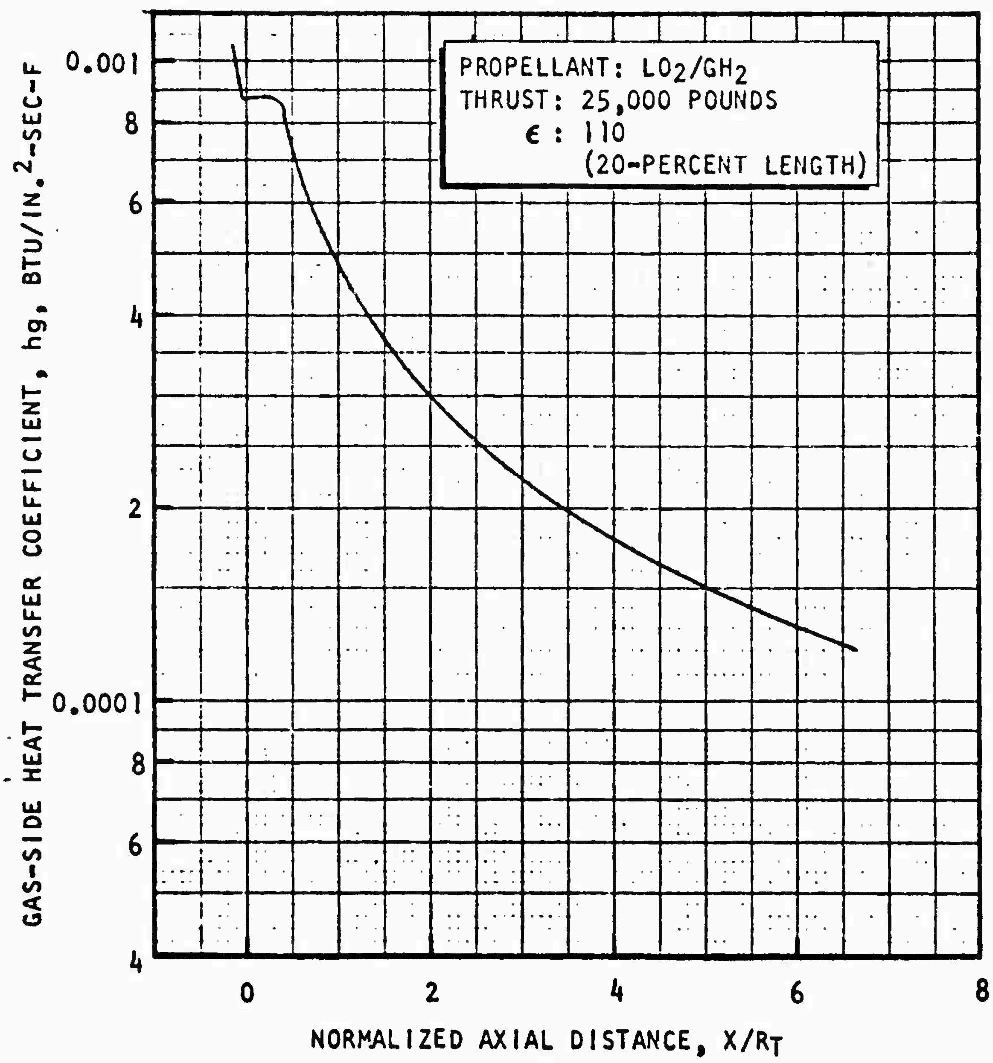


Figure 81. Single-Panel Nozzle Gas-Side Heat Transfer Coefficient Distribution (P_c = 750 psia, MR = 5.5)

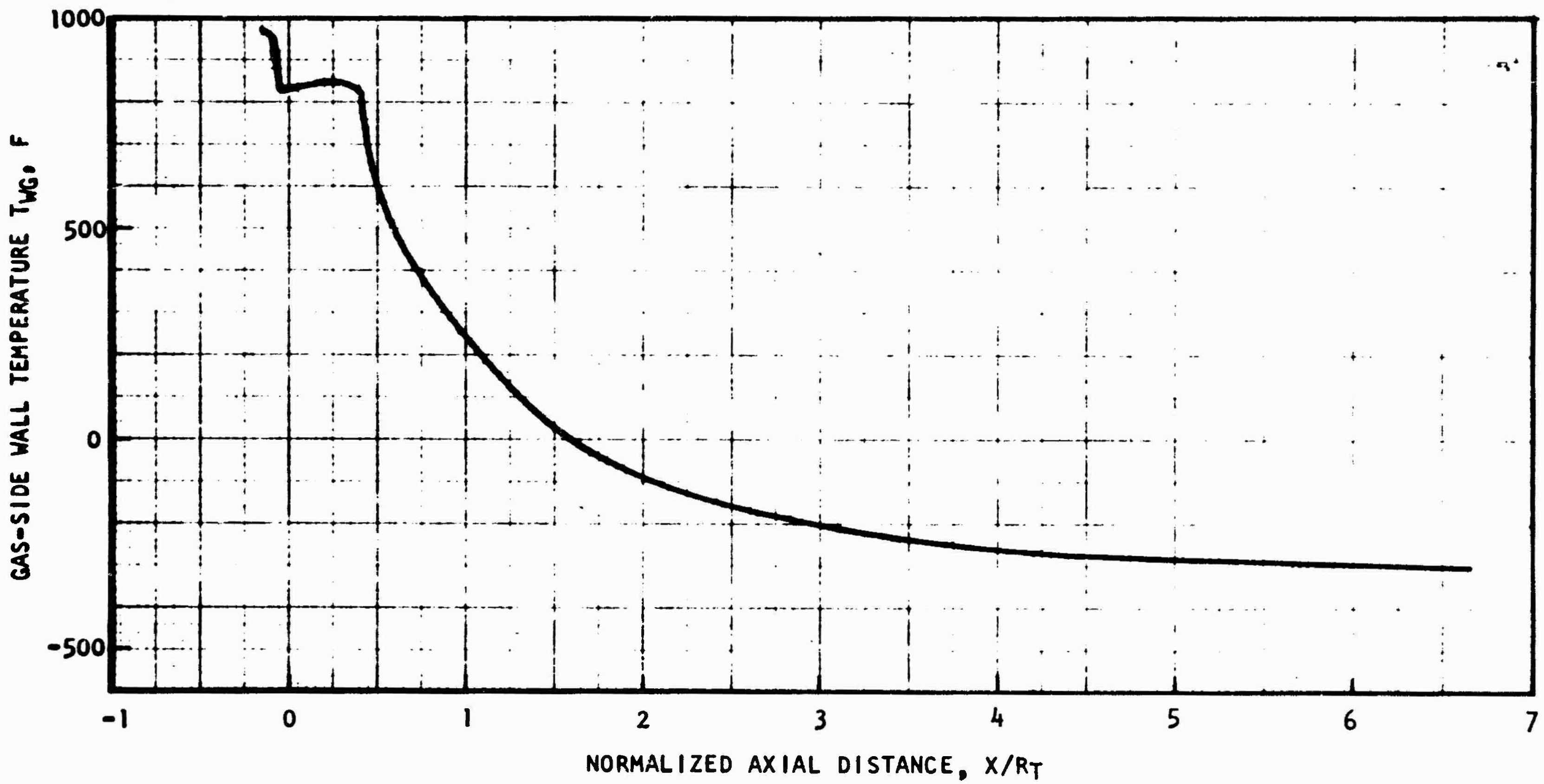


Figure 82. Single-Panel Nozzle Gas-Side Wall Temperature Distribution (P_c = 750 psia, MR = 5.5)

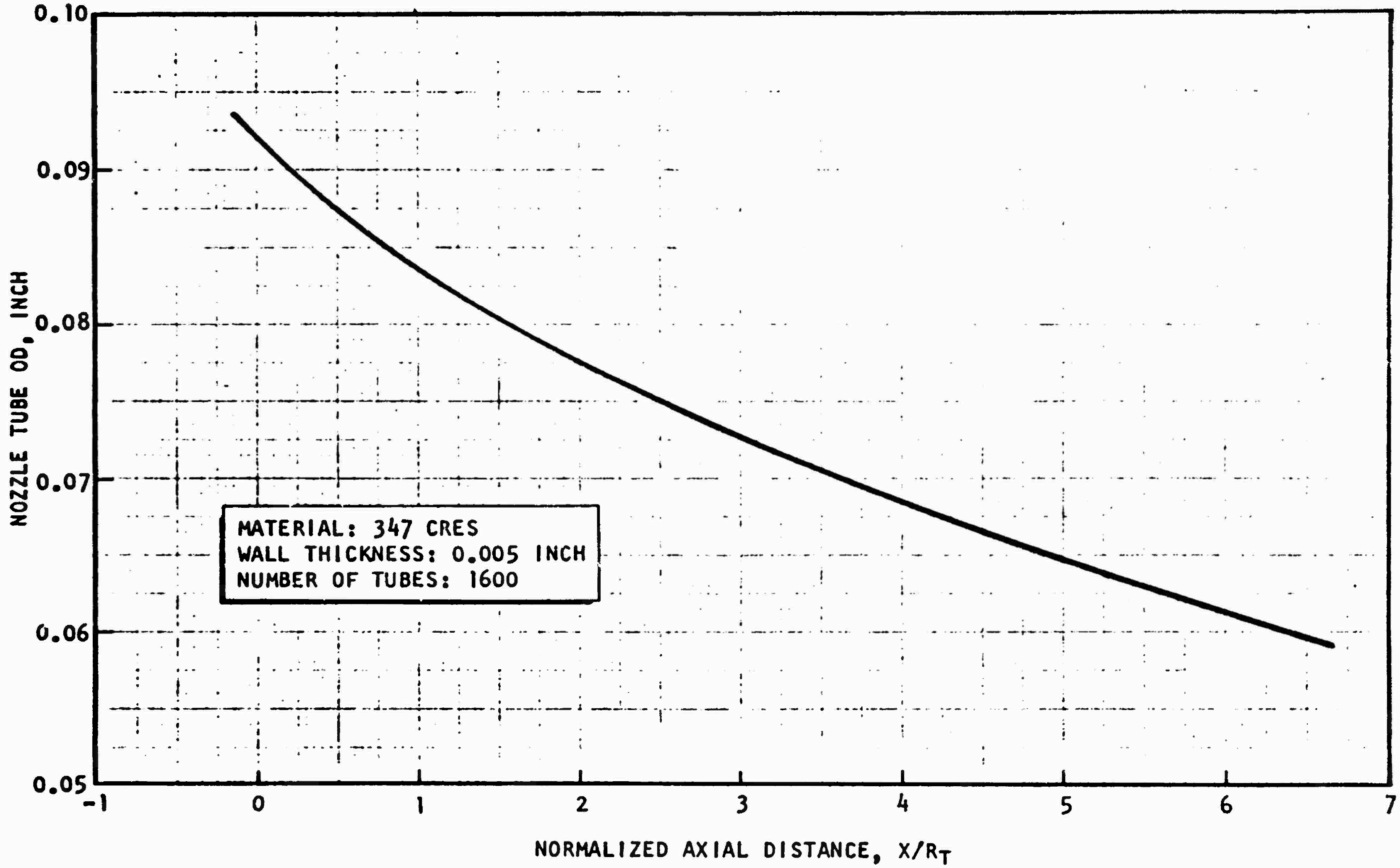


Figure 83. Single-Panel Nozzle Tube Dimensions

Base Closures

A base closure assembly is required for the thrust chamber assembly to:

1. Prevent the recirculation of hot, primary combustion exhaust gas into the center section of the aerospike thrust chamber
2. Permit the development of additional thrust by introduction of a small percentage of the turbine exhaust gas at the base plane for development of a positive base pressure

The base closures will consist of a fabricated assemblies with multiple orifices for discharge of the base flow. The assemblies will be weld attached to the nozzle extensions.

Thrust Mounts and Gimbals

The engine thrust mounts consist of 6 titanium tubular "A" frames mounted to 12 structural tie-bolts. The apex of the "A" frames is pinned to a cone compression member. At the top of the cone, a spherical rod end bearing serves as the engine gimbal. Different size structures are required for the single- and double-panel thrust chambers.

DOUBLE-PANEL ENGINE TURBOPUMPS

A design study was conducted to define conceptually the liquid hydrogen and liquid oxygen turbopumps for the AMPS 25,000-pound-thrust, 1000-psia chamber pressure engine. Final engine balance results, as presented earlier in this report, were obtained with the final component performance information and after the design layouts were completed.

Suction performance considerations and the objective of minimum size established the speed of the fuel turbopump at 75,000 rpm and the speed of the oxidizer turbopump at 22,000 rpm. Maintaining those speeds constant, tradeoff studies were

conducted, comparing performance, relative cost, and weight and engine power margin. The details of these tradeoff studies are presented in Appendix C.

Included in the comparison were two- and three-stage centrifugal fuel pumps with one- and two-row axial flow impulse turbines and a single-stage oxidizer pump with a single-row partial admission turbine. The potential of using the fuel turbine wheel design for the oxidizer turbine also was evaluated.

On the basis of studies presented in Appendix C, the fuel turbopump was established as a three-stage centrifugal pump with a single-row, full-admission axial flow impulse turbine. The oxidizer turbopump was defined as a single-stage centrifugal pump with a single-row partial admission axial turbine. The application of the fuel turbine wheel to the oxidizer turbine proved to be unfeasible.

LH_2 Turbopump Design

Fluid Dynamic Design. The final fluid dynamic analysis of the LH_2 turbopump was based on the requirements of the engine balance noted in Table 40.

The performance parameters and control dimensions of the LH_2 turbopumps are presented in Table 40. In Fig. 84, the pump head coefficient and efficiency are given as a function of flow coefficient. The expressions for the coordinates are normalized relative to design point values to permit the application of the curves directly in the engine balance computer program.

The operating speed of the liquid hydrogen pump (75,000 rpm) was established at the maximum attainable with the available NPSH. The speed selection was based on a suction specific speed of 99,000 which corresponds to an inlet total head of one velocity head ($C_m^2/2g$). The technology base for this criterion has been established on the J-2 and J-2S fuel pumps which have demonstrated satisfactory operation with one velocity head.

With the three-stage pump, the specific speed per stage obtained was 870, resulting in a pump efficiency of 62 percent. The head requirements were met with three 4.3-inch-diameter impellers, with a tip speed of 1408 ft/sec.

TABLE 40. LH_2 TURBOPUMP

PERFORMANCE PARAMETERS AND CRITICAL DIMENSIONS

PUMP:	
Number of Stages	3
Rotating Speed, rpm	75,000
Flowrate, lb/sec	8.164
Flowrate, gpm	814
Isentropic Head, ft.	97,000
Discharge Pressure, psia	3270
Impeller Tip Diameter, in.	4.3
Impeller Tip Speed, fps	1408
Specific Speed Per Stage	870
Efficiency, percent	62
Horsepower	2320
Minimum NPSH required, ft.	60
TURBINE:	
Inlet Temp., ^{o}R	1000
Inlet Press., psia	1977
Exhaust Temp., ^{o}R	899
Exhaust Press., psia	1190
Press. Ratio	1.661
Speed, rpm	75,000
Blade Speed, fps	1635
Velocity Ratio	.337
Efficiency, percent	68.7
Flowrate, lb/sec	5.05
Horsepower	2320
Admission, percent	100
Stress Factor, $AAN^2 \times 10^{-9}$	27.0

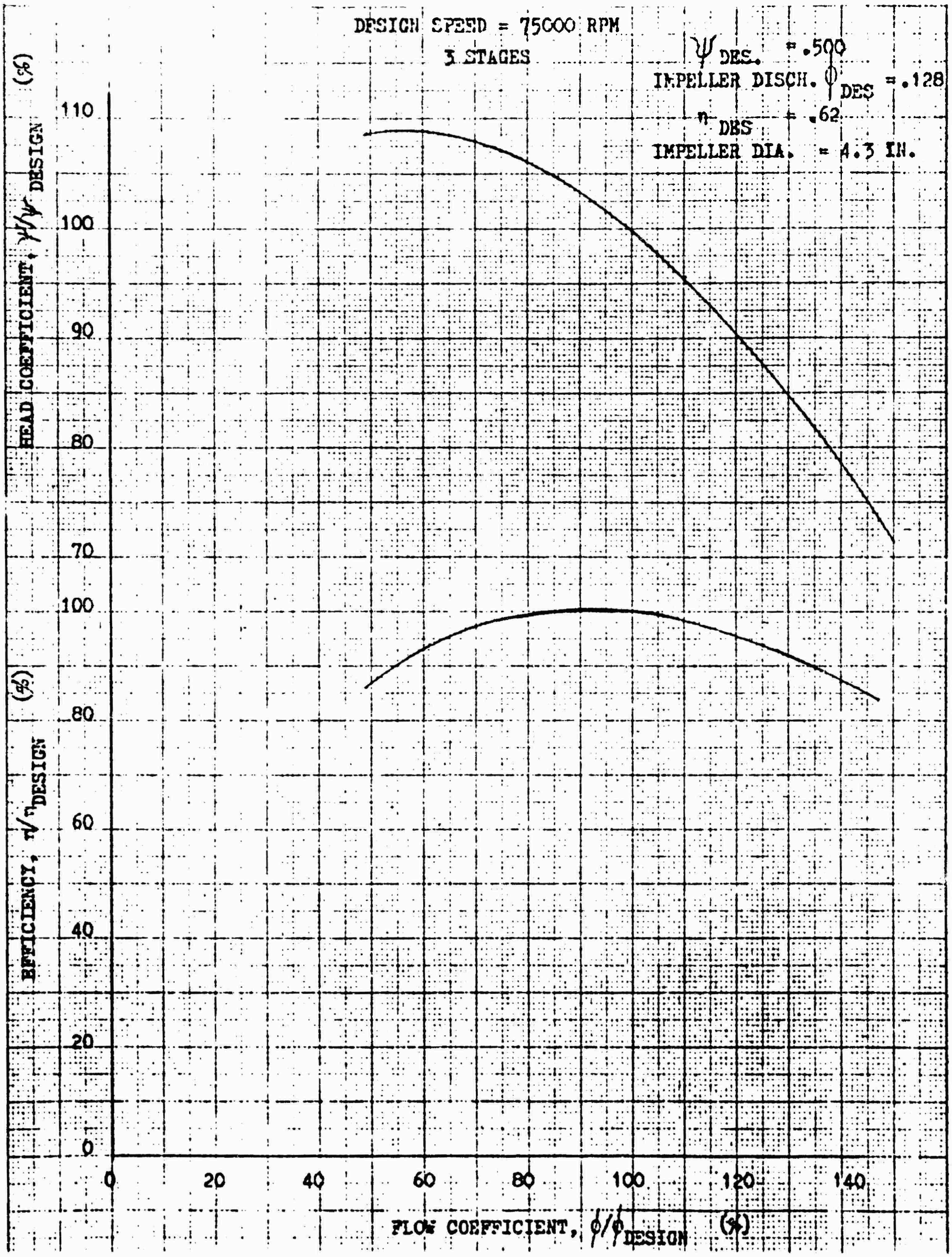


Figure 84. AMPS 25K Aerospike LH_2 Turbopump Performance Map

To meet the power requirement of the pump (2320 hp) at 75,000 rpm, a single-stage impulse turbine was designed with a wheel pitch diameter of 5.00 inches. The wheel tip speed- to-gas-spouting velocity ratio (U/C_o) obtained was 0.337, resulting in a turbine efficiency of 68.7 percent. A turbine propellant flowrate of 5.05 lb/sec was required to meet the nominal pump power requirements. The efficiency of the fuel turbine as a function of U/C_o ratio is presented in Fig. 85. In Fig. 86, turbine performance is defined for various pressure ratios by plotting a flow parameter (ordinate) versus a speed parameter (along the abscissa).

Configuration Description. In Fig. 87, the final layout of the LH_2 turbopump is presented which defines the internal detail configuration of the selected design.

The pumping elements consist of an axial flow inducer required to meet the NPSH requirements, followed by three centrifugal impellers. The hydrodynamic passages of the three impellers are identical, which allows machining them from identical castings into the various finished configurations, effecting tooling and production cost savings. The first- and second-stage impellers are followed by hydrodynamically identical crossover sections, consisting of radially outward flow diffuser vanes followed by a vaneless turning passage and radially inward flow, vaned diffuser. The second diffuser row in each stage introduces the same amount of whirl into the liquid as the inducer, to permit the use of identical impellers. The last impeller discharges into a radially outward flow, vaned diffuser from where the fluid is collected in a scroll-shaped volute and delivered through a single discharge pipe.

The external part of the front crossover is a barrel-shaped shell which forms a structurally efficient pressure vessel over the pump. The front crossover serves as the support for the front bearing and also retains the rear crossover by pressing it to the volute backplate through a Bearium* ring (balance piston high-pressure ring) which will yield sufficiently under preload to allow the front crossover to bottom on the volute near the retaining nut. The axial load on the rear crossover under operating pressure is toward the pump inlet, thus the Bearium

* Trade name for lead-bronze mixture

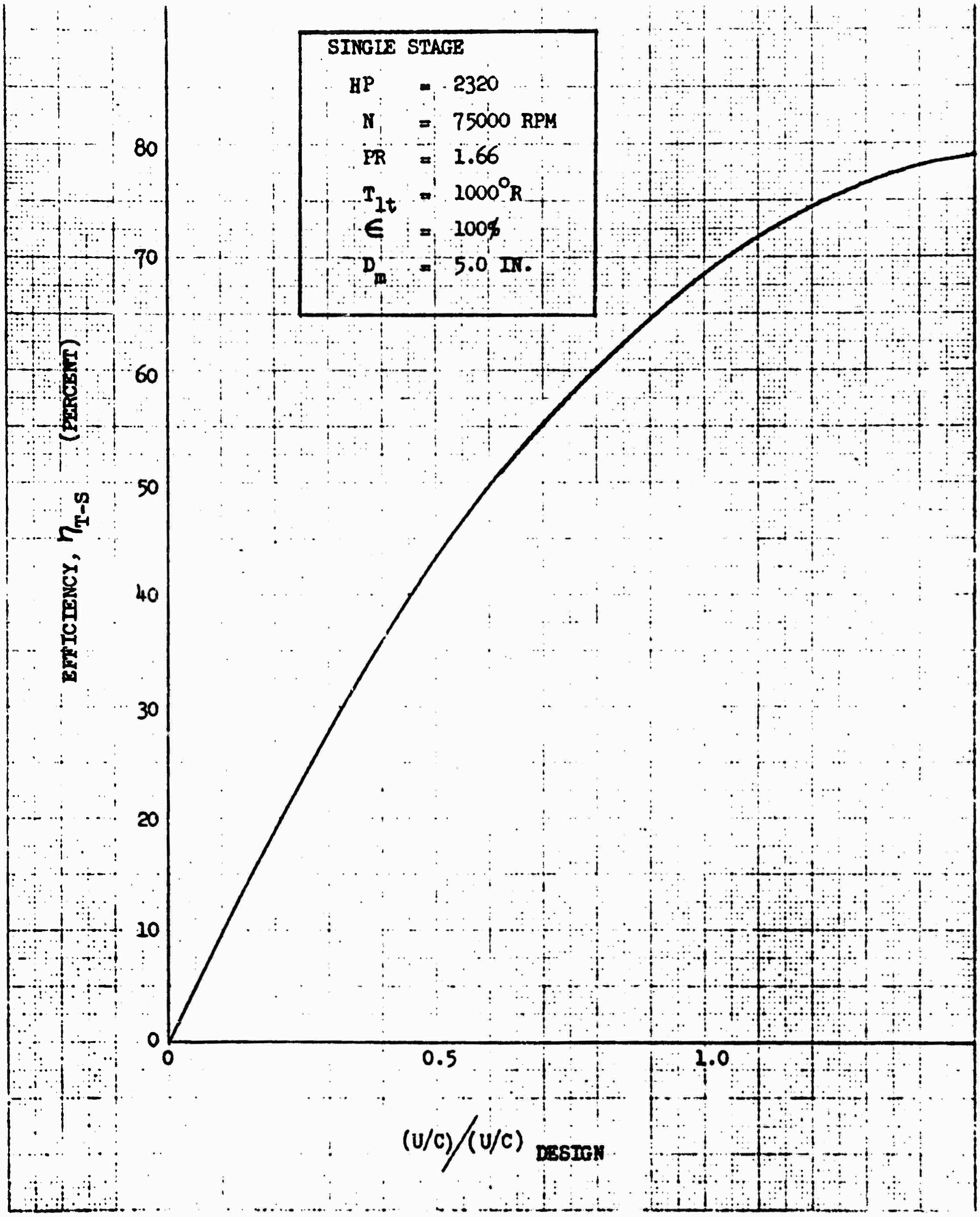


Figure 85. Amps 25K Aerospike Fuel Turbine Performance Map

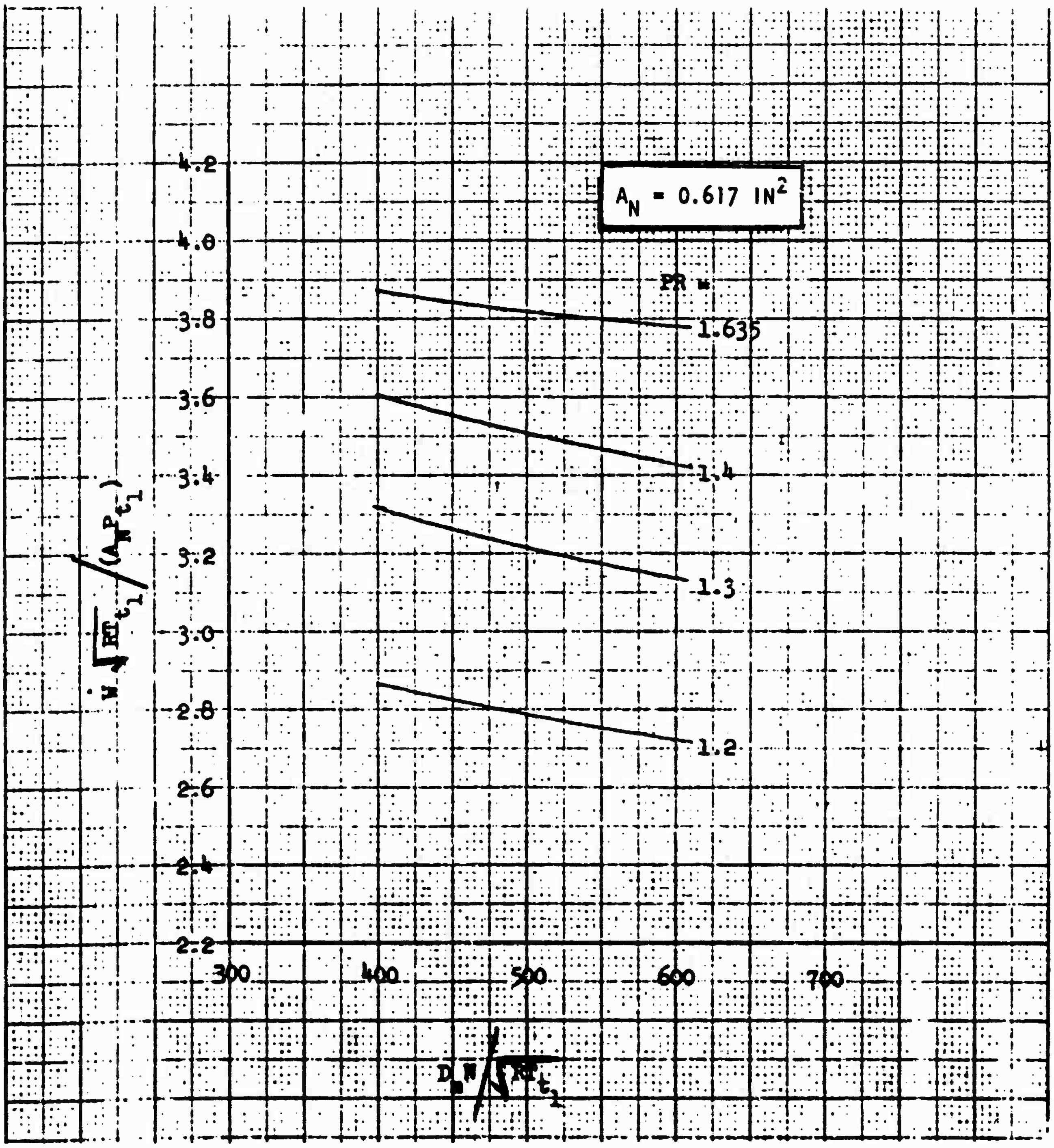


Figure 86. AMPS 25K Aerospike Fuel Turbine Performance Map

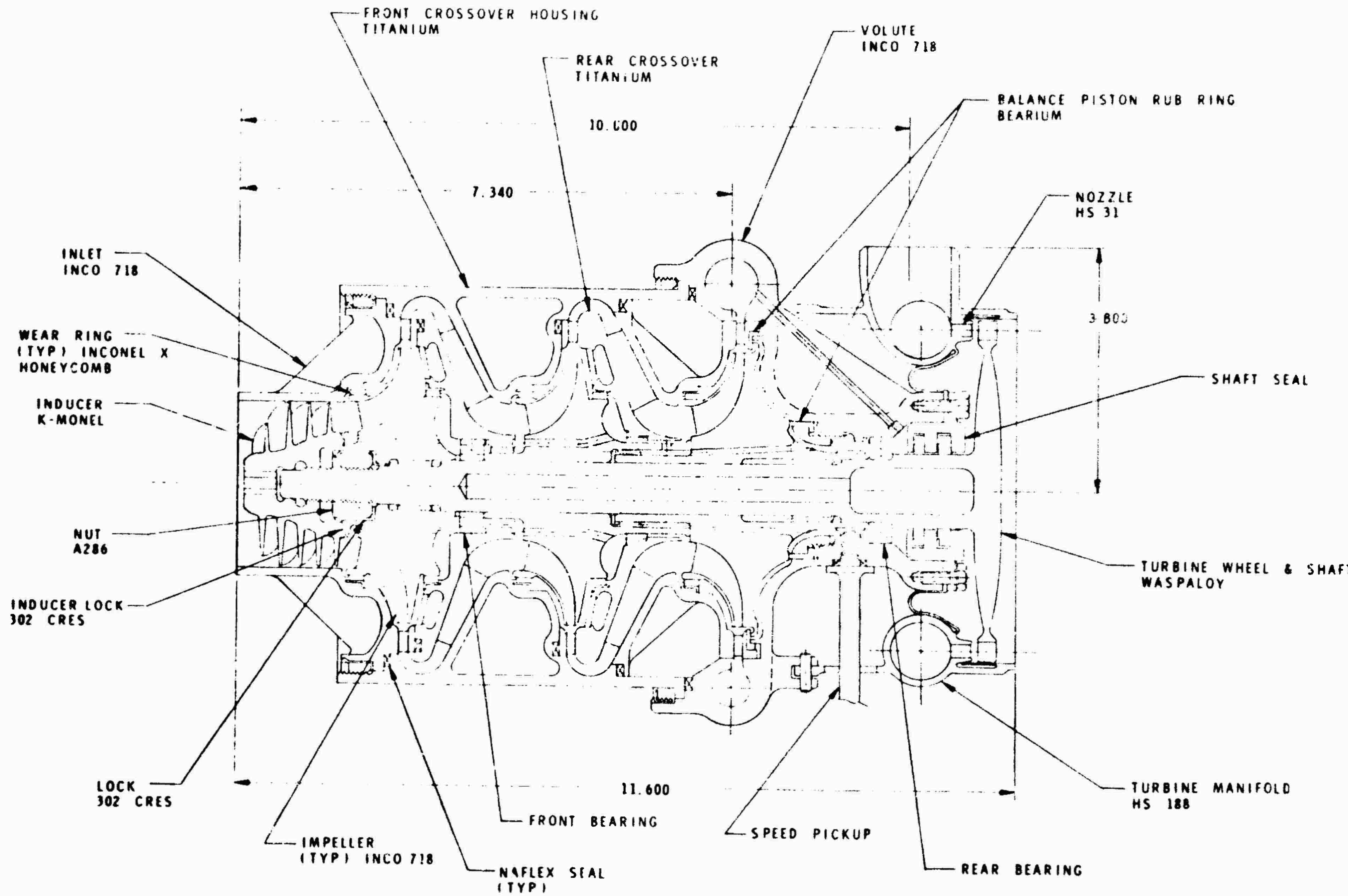


Figure 87. 25,000-Pound-Thrust Double-Panel Aerospike Engine Hydrogen Turbopump

ring is not subjected to further compression. Both crossover pieces could be cast from INCO 718, or welded from either Inconel or titanium. For the purposes of this conceptual study, a welded titanium construction was assumed. It is also within current state of the art to cast these parts from titanium, but some casting development effort would be required. This possibility should be reviewed at the time of the final detail design in the context of specific program objectives and schedules and in light of further advances in titanium casting technology.

The impeller tip speed required to generate the desired head was 1408 ft/sec, sufficiently low to permit using conventional INCO 718 castings for the impellers. The inducer is machined from a pancake forging. Of the several materials which could be used, K-monel offered the best compromise of high strength and machinability and, therefore, was selected. All three impellers, as well as the two bearing inner races, are axially preloaded against the shaft shoulder with a nut located forward of the first-stage impeller. Splines are used to transmit the torque from the shaft to the impellers. The inducer is threaded directly on the shaft and locked with a deformed lock-tab located between the inducer and the first-stage impeller. The thread direction will be such that the inertia effect during start, as well as the operational torque loads, will tend to tighten the inducer. This concept of retaining the inducer has been successfully demonstrated on Rocketdyne's Mark 36 liquid fluorine pump at speeds up to 104,000 rpm (NAS3-12022).

The volute that also will serve as the rear bearing support and shaft seal housing will be cast from INCO 718, as will the inlet to the pump. These two parts will be in-place welded to the engine inlet and discharge ducts, respectively, which will act as mounting points for the turbopump.

Shaft axial thrust control is effected by locating the wear rings behind the first- and second-row impellers at the proper diameters to obtain zero thrust. A self-compensating balance piston is incorporated into the third-stage impeller to absorb any variations in the axial loads. To operate the balance piston, high-pressure liquid hydrogen is bled from the discharge of the third-stage impeller

and passed through an orifice at the outer diameter of the impeller into the balance piston cavity. From the balance piston cavity, the fluid is returned through a low-pressure orifice into the eye of the impeller. If there is an unbalanced axial load toward the pump inlet, the rotor will move forward, closing the high-pressure orifice gap and opening the low-pressure orifice gap. The resulting decrease in the balance piston cavity pressure level will introduce a correcting axial force toward the turbine end. Conversely, if there is an unbalanced axial load toward the turbine, the rotor will move aft, opening the high-pressure orifice gap and closing the low-pressure orifice gap. As a result, the cavity pressure will increase until the unbalanced load is cancelled.

The turbine manifold is mounted on the pump volute through a radial pin joint which allows thermal differential expansion. The manifold will be fabricated by welding preformed details of Haynes-188 material. The nozzle will be cast as an integral piece of Haynes-31 and electron-beam welded into the manifold. The turbine disk and the shaft will be rough machined from Waspaloy forgings and welded together in the vicinity of the shaft rider seal. The turbine rotor blades will be machined integral with the disk.

Mounting of the turbopump to the engine will be accomplished by in-place welding rigid engine ducting to the pump inlet and turbine inlet.

Bearings. The pump rotor is supported on two angular contact ball bearings. The bearings were designed to fulfill the requirements for high-speed operation and long life. To minimize the detrimental speed-related phenomena, the bearing bores were minimized consistent with shaft strength and stiffness requirements. A digital computer program was used to determine bearing fatigue life and reactions, accounting for speed, loads, and bearing geometry. Consideration was given to using the duplex bearing arrangement shown in Fig. 88 because it offered a better control over bearing preloads and added to the radial load capability. However, a rotordynamic analysis disclosed that due to the increased radial stiffness of the duplex bearings, the critical speed was shifted into the lower

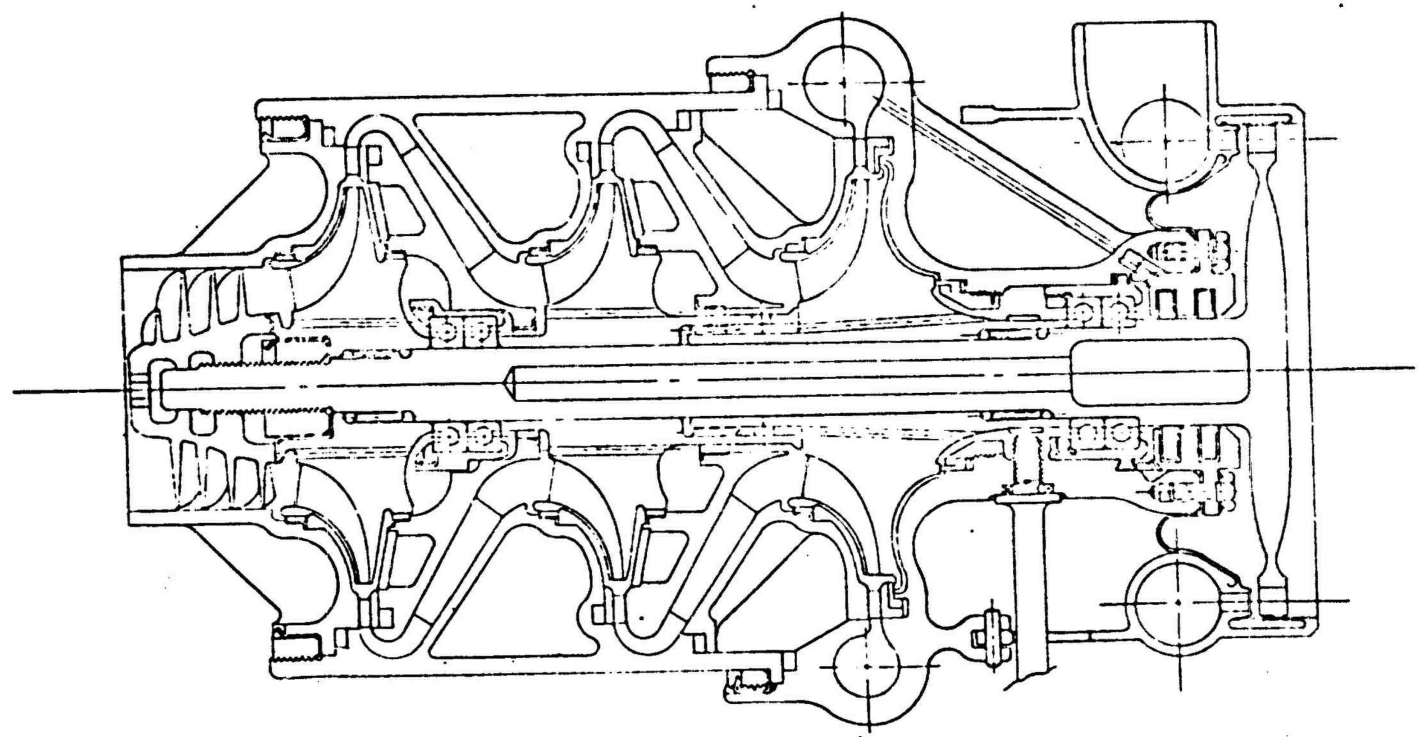


Figure 88. Double-Panel LH_2 Turbopump With Duplex Bearings

operating speed range of the turbopump. The duplex bearings could have been made acceptable by soft mounting, but it was believed that the additional complexity would offset the advantages offered by the duplex design.

Detailed dimensions of the bearings selected for the LH_2 turbopump are noted in Table 41 . Extra-extralight series bearings were chosen to minimize the ball sizes and inertia, therefore reducing the forces required to accelerate the balls.

The maximum DN value for LH_2 pump bearings (1.88×10^6) is within the present state of the art. The outer race curvatures were made closer than those of the inner race to compensate for the centrifugal loading encountered at high speed. The contact angles were made relatively low to enhance the radial stiffness. Because a balance piston was used to control axial loads, higher contact angles to accommodate large thrust loads were not necessary. Fatigue life versus load curves for the LH_2 pump bearings are shown in Fig. 89 and 90. A wide range of axial loads were investigated for the larger bearing because this bearing absorbs starting transient loads. The smaller bearing sustains preload only. The projected radial load on the basis of 0.1 gram-inch unbalance is 35 pounds at 75,000 rpm. Maximum transient axial loads are approximately 200 pounds. Adequate fatigue life was indicated for both bearings on the basis that new bearings are installed at overhaul (10-hour life).

The LH_2 bearing materials were chosen on the basis of prior experience with LH_2-cooled bearings. Races and balls are constructed of CEVM* 440-C stainless steel. All cages are one-piece Armalon (glass fabric-filled Teflon) with no external reinforcement.

Because of their higher speed, the LH_2 pump bearings were specified to a tolerance level per ABEC-7. AFBMA grade 10 balls were used to minimize ball size variation in new bearings.

---

*Consumable Electrode Vacuum melt

TABLE 41. DOUBLE-PANEL LH_2 BEARING DESIGN SUMMARY

POSITION	ENVELOPE DIM.				INTERNAL GEOMETRY						SPEED
	BASIC SIZE	BORE mm	O.D. mm	WIDTH mm	n	d in.	E in.	β_o deg.	F_i	F_o	
TURBINE END	1905	25	42	9	11	.21875	1.32	17	.53	.525	75,000
PUMP END	1904	20	37	9	9	.21875	1.12	17	.53	.525	75,000

n = number of balls

d = ball diameter

E = pitch diameter

β_o = initial contact angle

F = curvature $\left(\frac{\text{race radius}}{\text{ball diameter}}\right)$

i = inner

o = outer

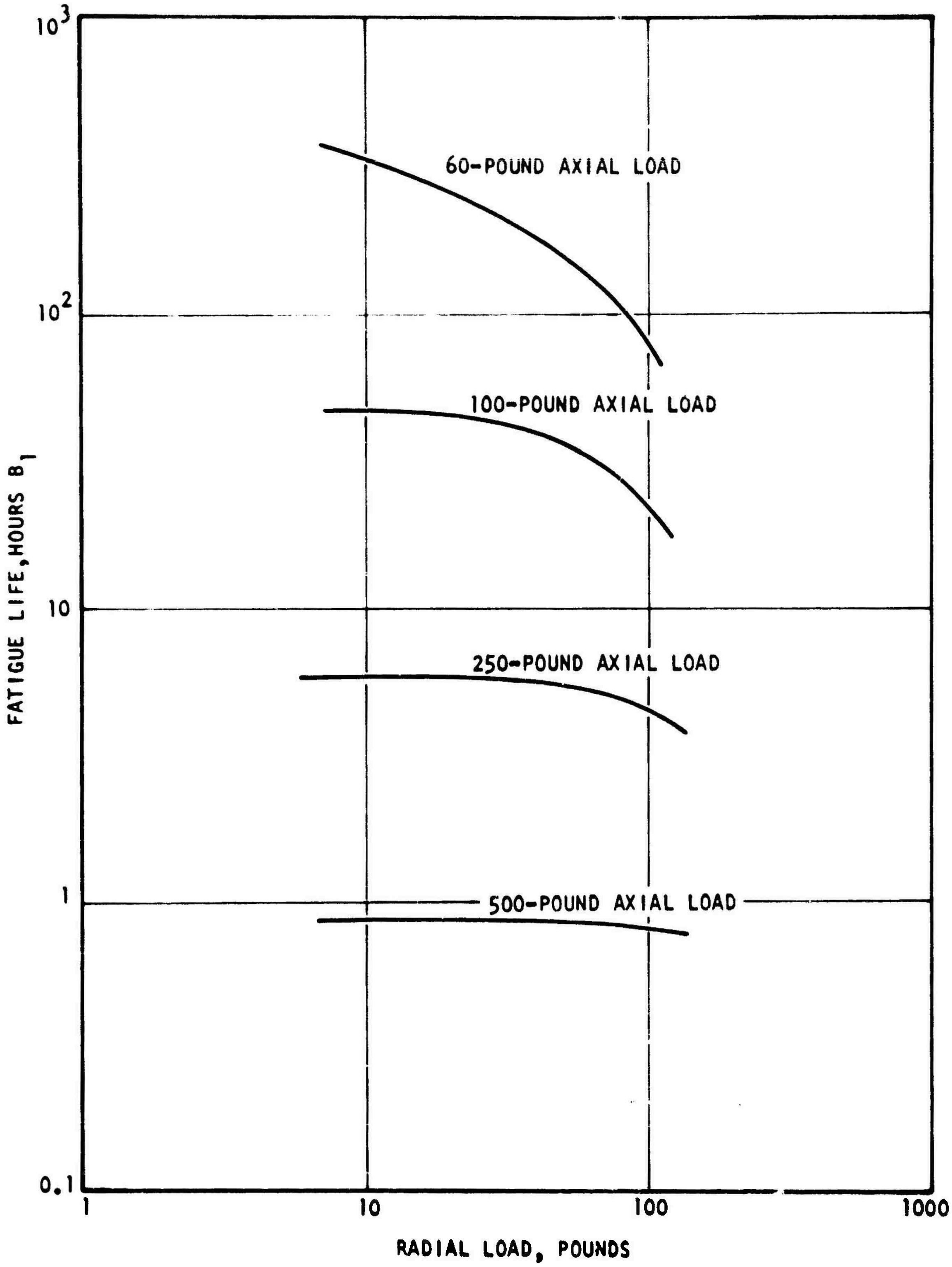


Figure 89. Double-Panel 1905 Size Ball Bearing Fatigue Life Versus Load at 75,000 rpm

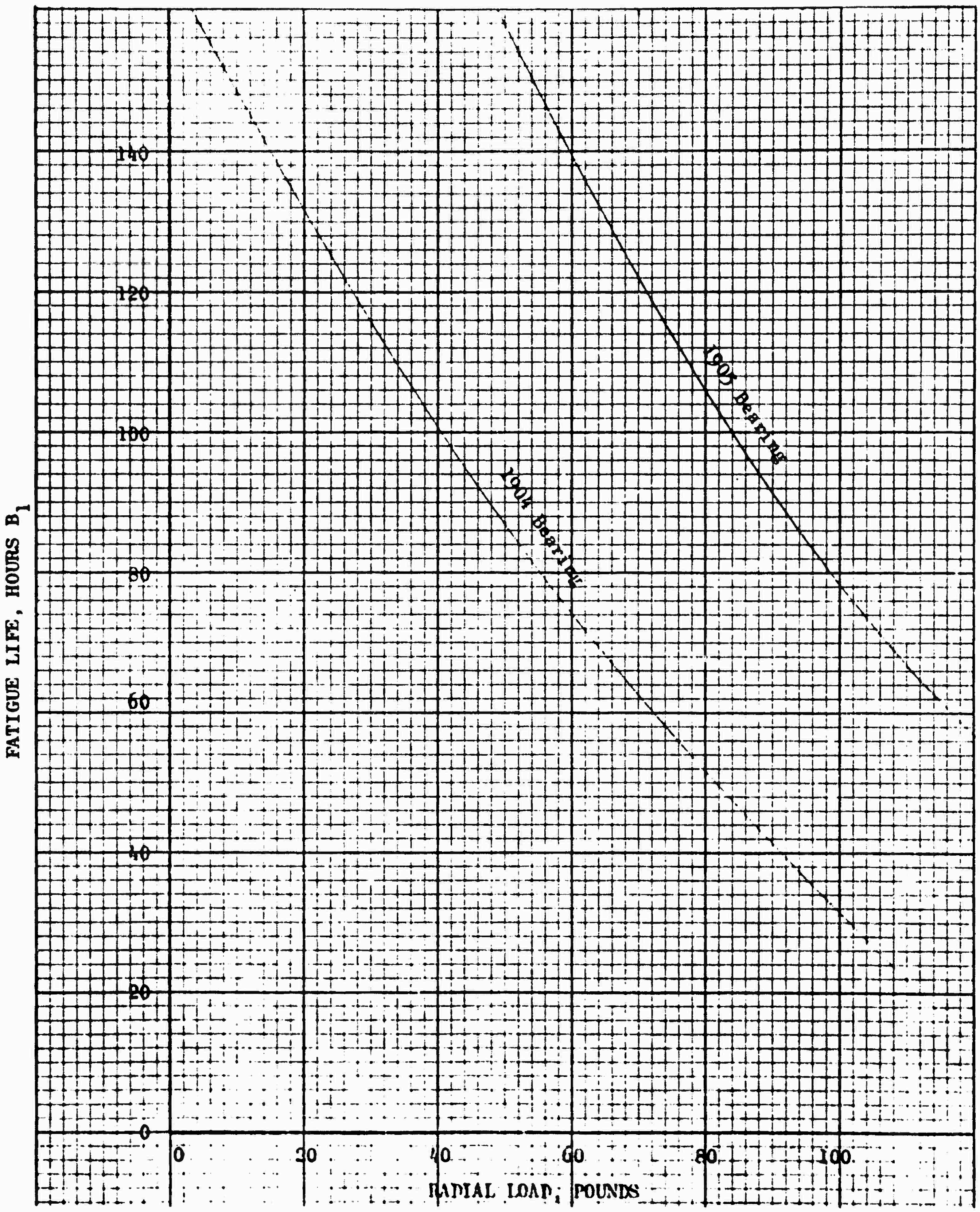


Figure 90. Double Panel, 1905, 1904 Size Ball Bearings Fatigue Life Versus Load at 75,000 rpm, 60-Pound Preload

Lubrication of the front bearing is accomplished by bleeding liquid hydrogen from the discharge of the first diffuser via a labyrinth seal and, after it passes through the bearing, allowing it to return to the main flow through axial holes in the first-stage impeller. To lubricate the rear bearing, liquid is tapped from the volute discharge and introduced on the turbine side of the bearing. From there, the liquid is passed through the bearing and returns to the inlet of the third-stage impeller through axial holes in the impeller hub.

Shaft Seals. Because the pump and turbine fluids (LH_2 and GH_2) are compatible, an absolute separation between the two regions was not mandatory. A valve located in the inlet line to the pump will be closed during coast periods; as a result, the seal package does not have to provide against loss of propellants during coast. Thus, the only function of the shaft seal is to minimize the leakage of LH_2 from the pump into the turbine. The predicted pressure on the pump side of the seal is 2000 psia, and 1200 psia on the turbine side. Thus, the flow direction will always be from the pump to the turbine.

Two configurations were considered to accomplish the flow restriction: a step labyrinth design and a double-floating, controlled-gap seal. While the labyrinth seal offered definite advantages with regard to simplicity, reliability, and cost, its calculated leakage rate was a substantial 0.21 lb/sec. In contrast, the leakage rate with the double control gap seal was 0.057 lb/sec. As a result, the latter concept was selected.

Rotordynamics. An analysis of the dynamic characteristics of the rotor was conducted. The critical speeds were calculated with a finite element method. The shaft was approximated as a series of concentrated masses and inertias connected by elastic beam elements. Forward synchronous precession was assumed and the bearings were modeled as linear springs to ground. The gyroscopic effect of each rotating mass was included.

The critical speed locations and trends are given in Fig. 91. The curves show that the rotor will operate between its second and third critical speed.

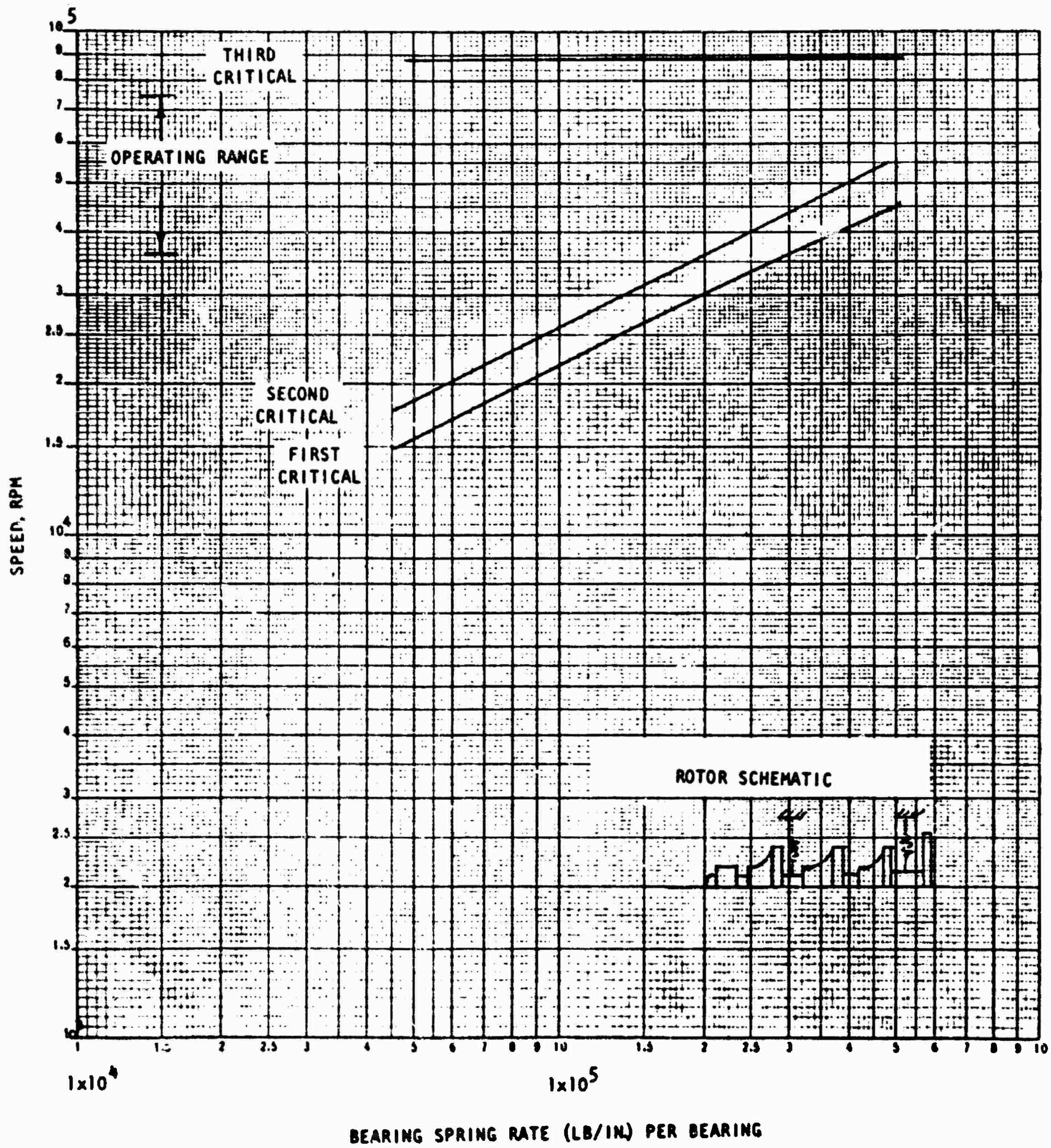


Figure 91. Double-Panel Fuel Turbopump Rotor Critical Speeds

As a result of the 5:1 throttling requirement of the engine, the turbopump must operate steady state anywhere between the nominal design speed of 75,000 rpm and the minimum throttling speed of 36,800 rpm. A 20-percent "pad" was applied at both ends of the above range as a safety margin. Figure 91 shows that, to keep the lower range free of critical speeds, the bearing spring rates are limited to approximately 100,000 lb/in. or smaller. A detailed analysis of the bearing and supporting structure is required to confirm compliance with this requirement. As part of the detailed design, the rotor will be made slightly stiffer to provide additional margin between the upper speed range and the third critical speed.

Stress. The fuel turbopump was sized to satisfy the structural criteria noted below:

Parameters	Required Factor of Safety
Ultimate Strength	1.4
Burst Pressure	1.5
High-Cycle Fatigue*	10 on Cycles
Low-Cycle Fatigue*	4 on Cycles
Creep Rupture*	4 on Time

*Accumulative damage functions

ϕ_{HCF} = High-Cycle Damage Fraction

ϕ_{LCF} = Low-Cycle Damage Fraction

ϕ_{R} = Creep Rupture Damage Fraction

$$10\ \phi_{HCF} + 4\ \phi_{LCF} + 4\ \phi_{R} = 1$$

The pump impellers are cast INCO 718 material. The minimum burst speed of the impellers is 95,000 rpm, providing an allowable operating speed of 80,000 rpm. The stage crossovers were fabricated from titanium, thereby providing a lightweight structure. The rear titanium crossover was used as a pressure-augmented

spring which provides a force to seal the diffuser vanes against the balance piston rub ring retainer. The pump inlet and discharge volute were cast INCO 718 material. The volute material was thermally compatible with the third-stage impeller, thereby providing minimum relative thermal deflections in the balance piston area.

The pump shaft and turbine wheel are integral and made of Waspaloy material. The coefficient of thermal contraction of the Waspaloy shaft matches that of the INCO 718 impellers, thereby minimizing piloting and axial stack problems. The shaft critical section occurs at the third-stage impeller shaft spline runout. The shaft factor of safety was determined utilizing the modified Soderberg equation which is based on the maximum shear failure theory. The minimum shaft factor of safety was 1.8.

The turbine manifold was fabricated from Haynes 188 material to afford maximum protection against the hydrogen environment embrittlement. The nozzle was cast Haynes 31 material.

Weight. The weight of the fuel turbopump was estimated on the basis of the cross sections shown in the layout of Fig. 87. A breakdown of the estimated weights is included in Table 42. The total weight of the turbopump was estimated at 39 pounds.

Maintainability. Inspection of the fuel turbopump will be performed after 2 hours of operation or 60 starts. Included in the inspections will be visual examination of the turbine and bearing through the speed pickup access port with a fiber optic tool. The turbine rotor blades and nozzle vanes will be examined through instrumentation ports using fiber optics and the integrity of the shaft may be verified by performing a gas leak check.

TABLE 42. DOUBLE-PANEL FUEL TURBOPUMP WEIGHTS

	Pounds
Inducer	0.5
Impellers (3)	7.8
Shaft	1.4
Wheel	1.6
Miscellaneous Parts	1.0
Total Rotating Parts	12.3
Inlet Housing	3.9
Intermediate Housing	8.7
Volute	10.6
Manifold	3.5
Total Static Parts	26.7
Total Fuel Turbopump	39.0

After 10 hours of operation or 300 starts, the turbopump will be completely disassembled and the bearings, as well as the shaft seal, balance piston seals, and the impeller seals, will be replaced. During this procedure, all parts will be visually examined and all rotating parts and pressure vessels will be dye-penetrant inspected to guard against incipient cracks or other signs of degradation.

Double-Panel LOX Turbopump Design

Fluid Dynamic Design. The performance parameters and control dimensions of the LOX turbopump are presented in Table 43. In Fig. 92, the pump head coefficient and efficiency are given as a function of flow coefficient. The expressions for the coordinates are normalized relative to design point values to permit the application of the curves directly in the engine balance computer program.

TABLE 43. DOUBLE-PANEL LOX TURBOPUMP PERFORMANCE PARAMETERS AND CRITICAL DIMENSIONS

PUMP:	
Number of Stages	1
Rotating Speed, rpm	22000
Flowrate, lb/sec	44.903
Flowrate, gpm	283
Isentropic Head, ft.	3420
Discharge Pressure, psiat	1716
Impeller Tip Diameter, in.	4.9
Impeller Tip Speed, fps	469
Specific Speed Per Stage	825
Efficiency, percent	66.5
Horsepower	420
Minimum NPSH Required, ft.	16
TURBINE:	
Inlet Temperature, ^{o}R	1001
Inlet Pressure, psia	1712
Exhaust Temperature, ^{o}R	951
Exhaust Pressure, psia	1195
Pressure Ratio	1.43
Speed, rpm	22000
Blade Speed, fps	768
Velocity Ratio	.187
Efficiency, percent	39
Flowrate, lb/sec	2.24
Horsepower	420
Admission, percent	20
Stress, Factor, $AAN^2 \times 10^{-9}$	4.66

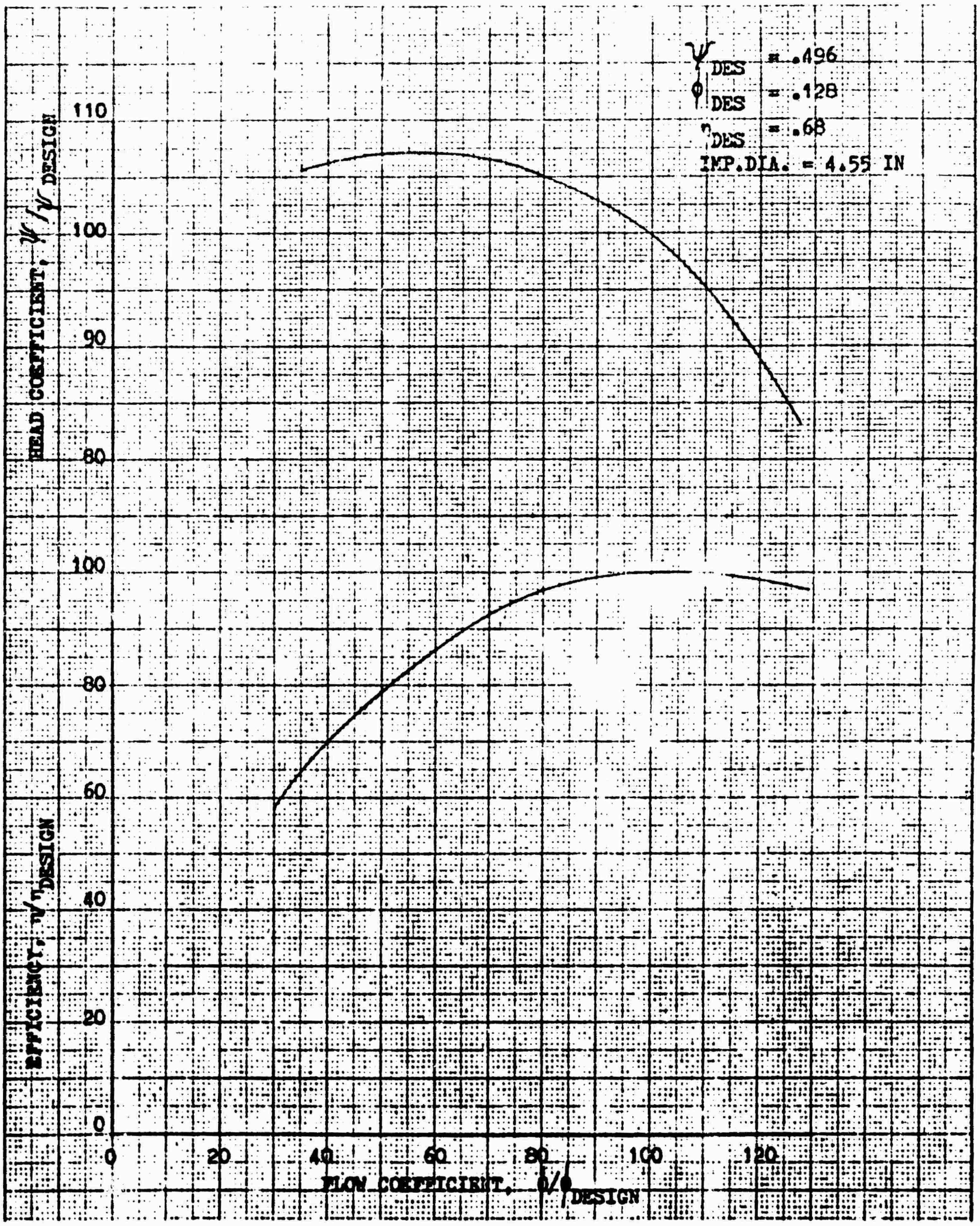


Figure 92. Double-Panel AMPS 25K Aerospike LOX Pump Performance Map, Design Speed = 22000 rpm, 1 Stage

The nominal operating speed of the liquid oxygen pump (22,000 rpm) was established at the maximum attainable with the available NPSH. The speed selection was based on a suction specific speed of 46,000 which corresponds to an inlet condition of 2.5 velocity heads ($C_m^2/2g$). The technology base for this criteria has been established on the J-2 and J-2S oxidizer pumps which have demonstrated satisfactory operation at these levels.

The design specific speed is 825, resulting in a pump efficiency of 66.5 percent. The head requirements were met with a 4.9-inch-diameter impeller, with a tip speed of 469 ft/sec.

To meet the power requirement of the pump (420 hp) at 22,000 rpm, a partial-admission, single-stage impulse turbine was designed with a wheel pitch diameter of 8.00 inches. The wheel tip speed-to-gas spouting velocity ratio (U/C_o) obtained was 0.187, resulting in a predicted turbine efficiency of 39 percent. A turbine propellant flowrate of 2.24 lb/sec was required to meet the nominal pump power requirements. The efficiency of the fuel turbine as a function of U/C_o ratio is presented in Fig. 93. In Fig. 94, turbine performance is defined for various pressure ratios by plotting a flow parameter (ordinate) versus a speed parameter (along the abscissa).

Configuration Description. In Fig. 95, the final layout of the LOX turbopump is presented which defines the internal detail configuration of the selected design.

The pumping elements consist of an axial flow inducer required to meet the NPSH requirements, followed by a centrifugal impeller. Fluid from the impeller is discharged into a vaned radial diffuser, collected in a scroll-shaped volute, and delivered through a single discharge pipe.

The inducer is machined from a K-monel forging which offers an excellent combination of strength and resistance to chemical reaction in LOX. The inducer is threaded directly on the shaft and secured with a sheet metal lock-tab

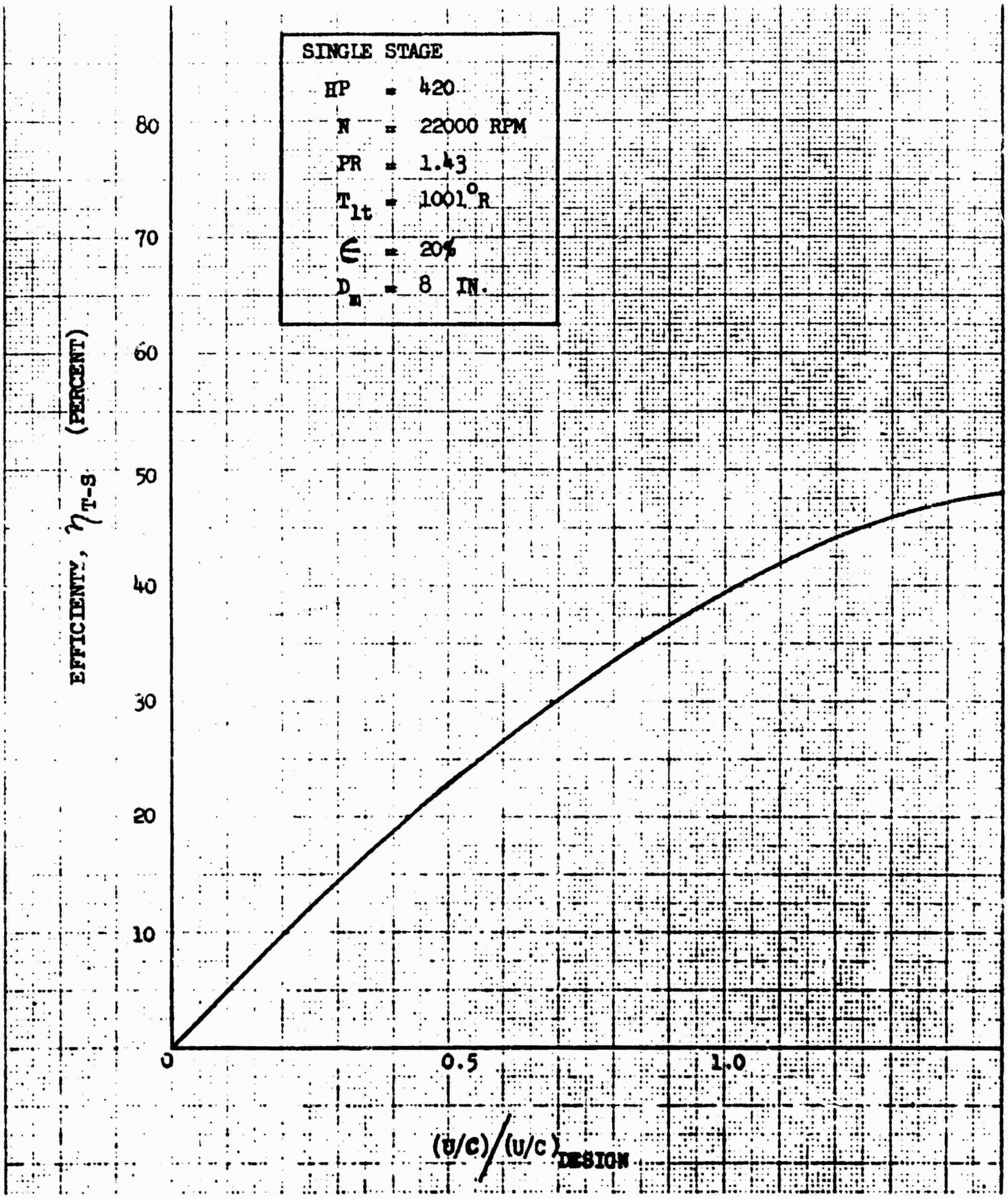


Figure 93. Double-Panel AMPS 25K Aerospike Oxidizer Turbine Performance Map

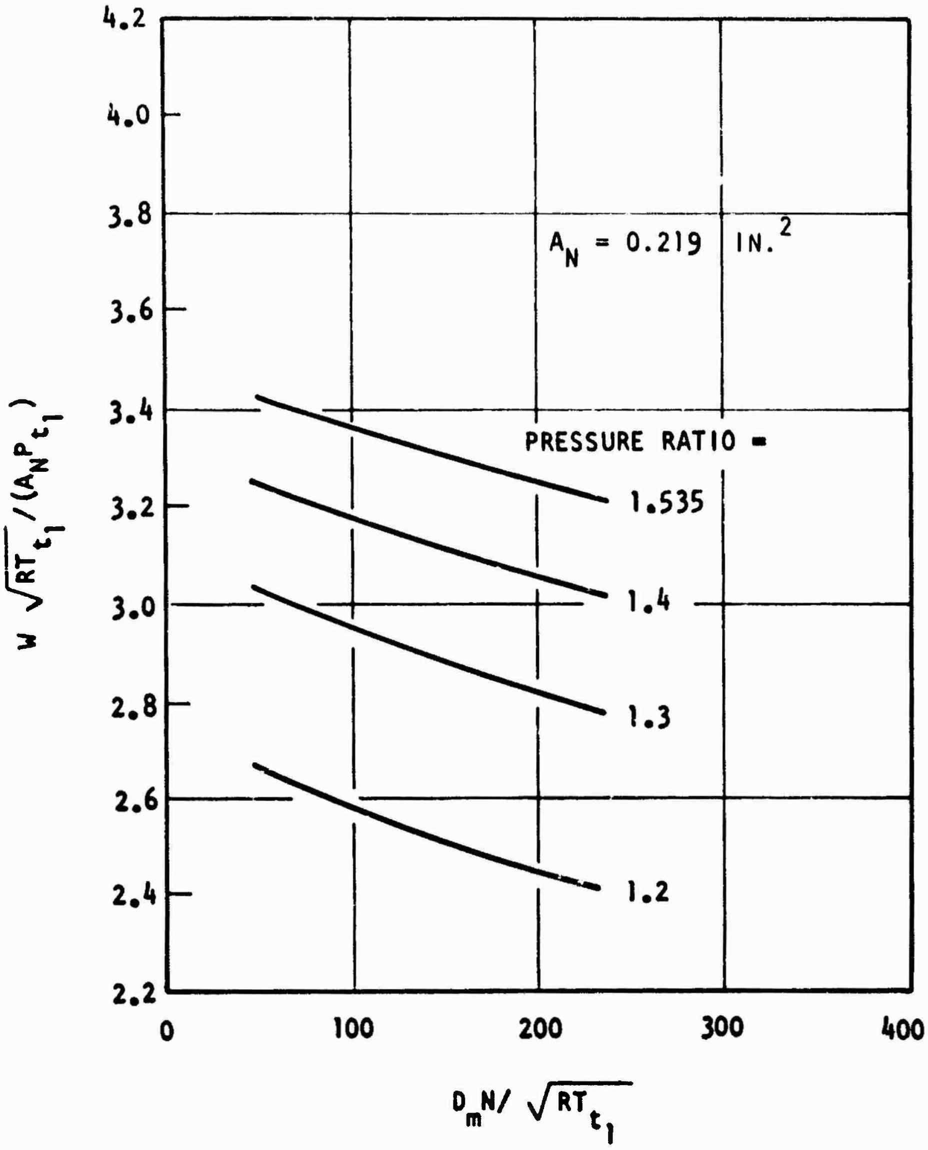


Figure 94. AMPS 25K Aerospike Double-Panel Oxidizer Turbopump Performance Map

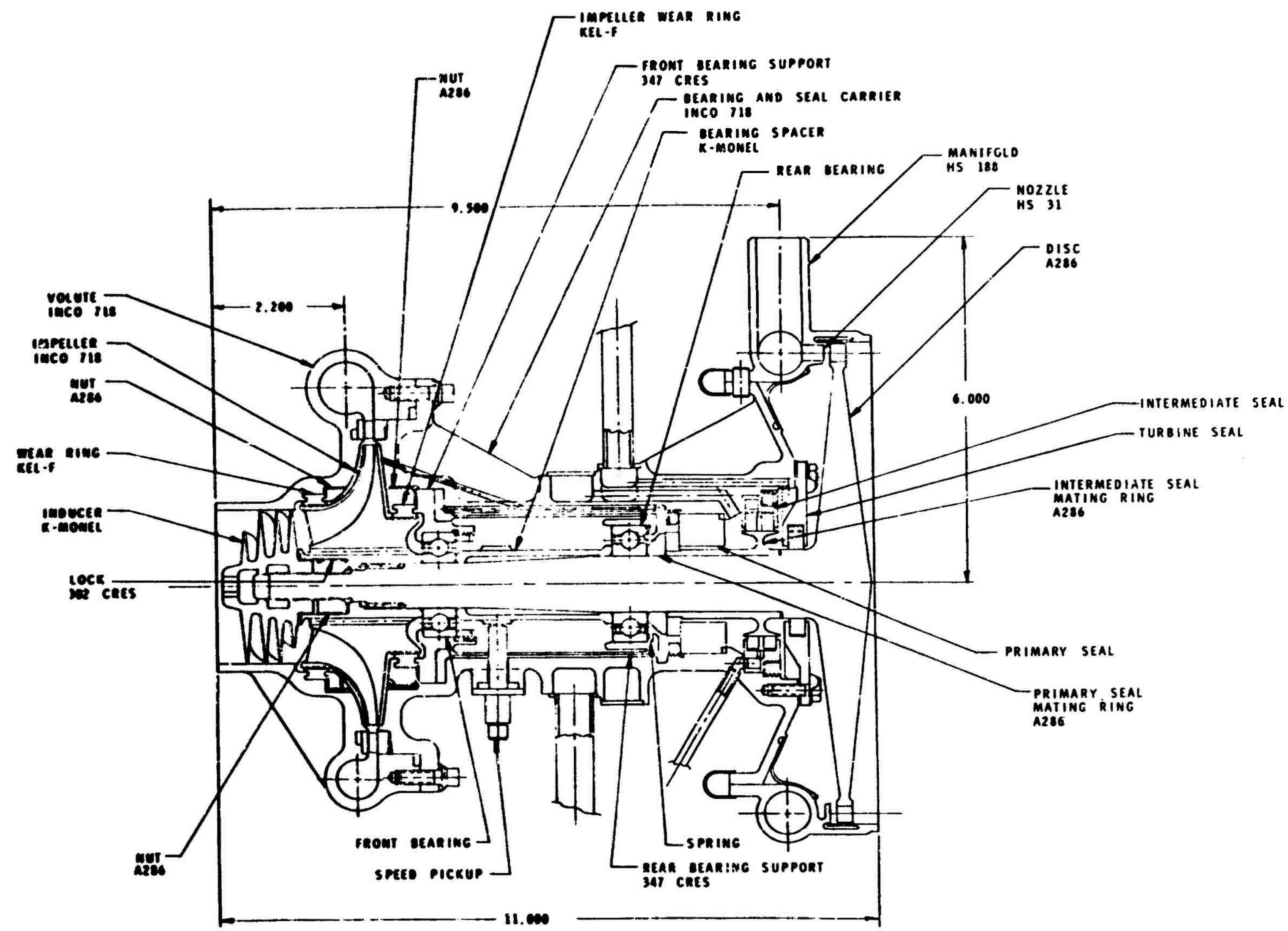


Figure 95. 25,000-Pound-Thrust Double-Panel Aerospike Engine Oxygen Turbopump

located between the inducer and impeller. The thread direction is such that the inertia effect during start as well as the operational torque loads tend to tighten the inducer.

The impellers, volute, and the diffuser/bearing carrier subassembly are cast from INCO 718. To facilitate casting and to obtain better tolerance control, the diffuser is cast separately and joined to the bearing carrier by welding. The volute is in-place welded to the engine inlet and discharge ducts. These joints also act as the mounting points for the turbopump.

Shaft axial thrust control is effected by locating the impeller rear wear ring diameter so that the net axial thrust becomes zero. Any residual thrust due to calculation error and off-design operation is carried by the front bearings. Kel-F is used for both impeller wear rings, allowing close wear ring clearances which are desirable from a performance standpoint, without the hazards presented by metal-to-metal contact.

Rotordynamic considerations dictated a low radial spring constant for the turbine end bearing. To comply with this requirement, the bearing was "soft-mounted," i.e., the bearing support was made a flexible tubular cross section, cantilevered from the area of the front bearing. Excessive radial motion of the rear bearing was avoided by providing only a small clearance between the bearing support OD and the bearing carrier ID.

The turbine manifold is mounted on the bearing carrier flange through a pin joint which will allow radial growth of the manifold without introducing appreciable thermal stresses. External leakage through the pin joint is prevented by a welded sheet metal seal on the inside of the joint. An external cover is welded on the joint to prevent moisture from collecting and freezing between the joint and the internal seal. The manifold is fabricated by welding preformed details of Haynes-31, which are then electron-beam welded into the manifold.

The turbine disk and the shaft are rough machined from A-286 forgings and joined by welding near the disk hub. The turbine rotor blades are machined integral with the disk.

Bearings. The pump rotor is supported on two Conrad-type ball-bearings of 20-mm bore at the pump end and 25-mm bore on the turbine end. These bearings are a deep-groove type to withstand axial loads in either direction as well as radial loads. To minimize the detrimental speed-related phenomena, the bearing bores were minimized consistent with shaft strength and stiffness requirements. A digital computer program was used to determine bearing fatigue life and reactions, accounting for speed, loads, and bearing geometry.

Detailed dimensions of the bearings selected for the LOX turbopump are noted in Table 44 .

The LO_2 pump bearings operate at conservative DN values (0.55×10^6 max). The larger cross-section 200 series bearings were used to obtain their inherent added capacity. The contact angle of the thrust bearing was set at 25 degrees to increase thrust capacity. A contact angle of 17 degrees was chosen for the radial bearing because axial capacity is not a primary goal. The smaller contact angle increases radial stiffness and decreases the detrimental effects of gyroscopic moments, reducing ball spinning and wear.

The bearings are made of materials chosen on the basis of prior experience with LOX-cooled bearings. Races and balls are constructed of CEVM 440-C stainless steel. All cages are one-piece Armalon (glass fabric-filled Teflon) with no external reinforcement.

The LOX pump bearings are made to ABEC-5 tolerances or better. AFBMA grade 10 balls are used to minimize ball size variation.

TABLE 44. DOUBLE-PANEL LOX BEARING DESIGN SUMMARY

POSITION	ENVELOPE DIM.				INTERNAL GEOMETRY						SPEED
	Basic Size	Bore mm	O.D. mm	Width mm	n	d in.	E in.	β_o deg	F_i	F_o	
LO_2 Pump (Turbine end)	205	25	52	15	11	.28125	1.515	17	.52	.53	22000
LO_2 Pump (Pump end)	204	20	47	14	10	.28125	1.36	25	.52	.53	22000

n = Number of balls

d = Ball diameter

E = pitch diameter

β_o = Initial contact angle

F = Curvature $\left(\frac{\text{race radius}}{\text{ball diameter}}\right)$

i = Inner

o = Outer

The bearing life as a function of loads is shown in Fig. 96 and 97. The 204 bearing response to a wide range of axial loads was investigated, as this bearing sustains rotor axial loads in the turbopump. The response of the 205 bearing was determined for the preload of 60 pounds, as this bearing will experience only preload and radial loads in the turbopump. Radial loads were calculated at less than 10 pounds on the basis of 0.1 gram-inch rotor unbalance. Residual axial loads are expected to be less than 250 pounds. Figures 96 and 97 indicate ample life margin based on the projected loads for the bearing races and balls.

Another limiting factor on the life of the bearings is the wear rate of the Armalon cage. Although LOX-lubricated bearings have accumulated over 15 hours of operation, the number of starts has been limited to under 100. Despite the fatigue life margin indicated in Fig. 96 and 97, the bearings are planned to be replaced after 10 hours of operation or 300 starts, based on the expected life capability of the Armalon cages.

Shaft Seals. The pump and turbine fluids, LOX and GH_2, are not compatible chemically; therefore, they must be prevented from mixing. A face-type hydrodynamic pump seal, a purged intermediate seal and floating-ring, controlled-gap turbine seal were used to provide absolute separation between the two areas.

The hydrodynamic seal concept was selected for the pump primary seal because conventional face rubbing seals will not satisfy life and start requirements. Leakage of this primary seal is estimated to be 0.04 lb/sec of engine operation. A control gap floating ring seal also could be used, but it has the disadvantage of permitting higher leakage.

The feasibility of hydrodynamic seals has been demonstrated in component test programs; specifically, a Borg-Warner type seal has accumulated 14 hours of operation. Cycle life capability of this seal is expected to be high, but experience is required. Additional development effort is planned on contract NAS8-27759, so that by the time detail design is initiated, this seal concept should be well-established state of the art.

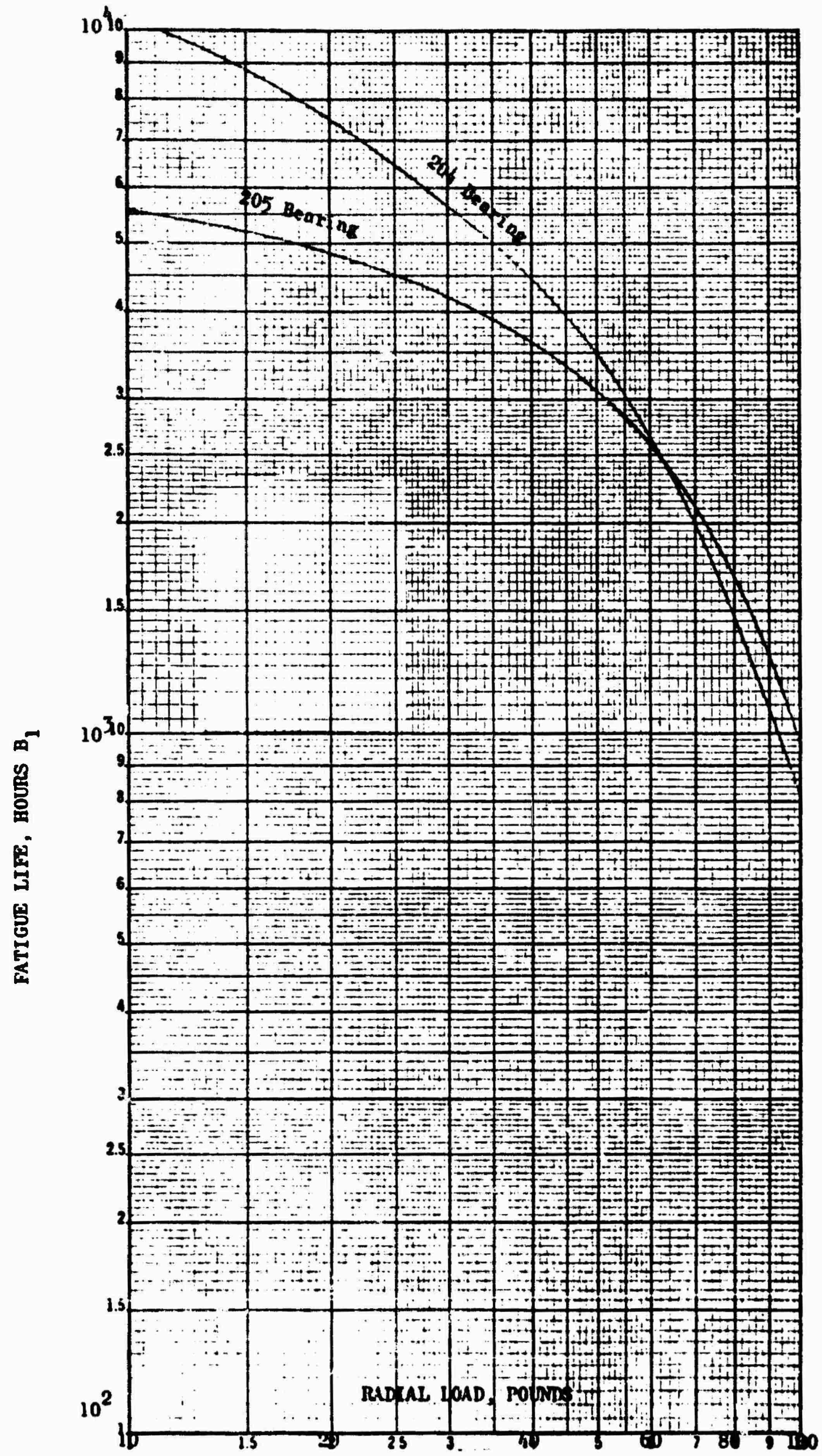


Figure 96. Double-Panel Fatigue Life vs Radial Load (204, 205 size ball bearings, 22,000 rpm, 60-pound reload)

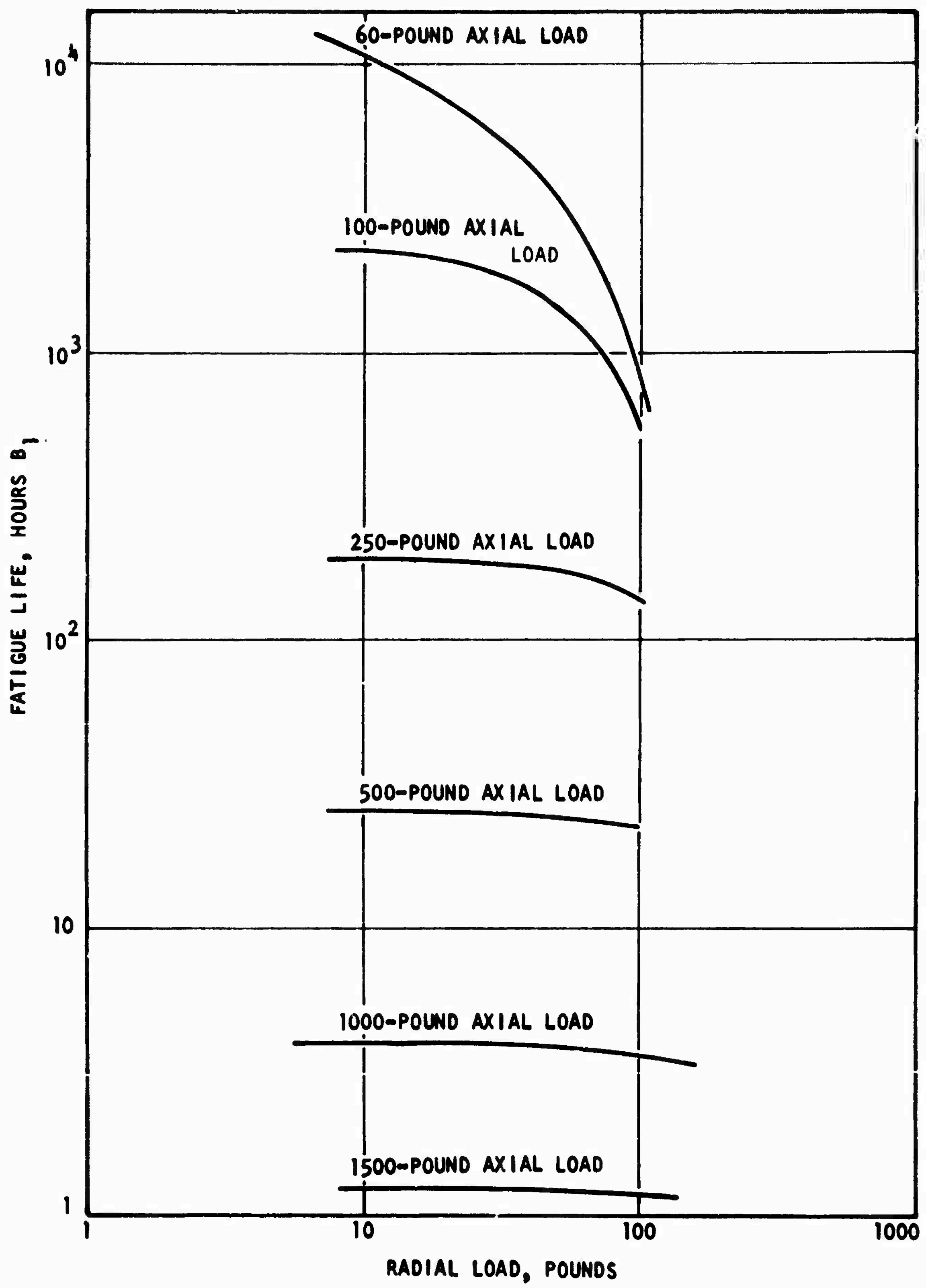


Figure 97. 204 Size Ball Bearing Double-Panel Fatigue Life vs Load at 22,000 rpm

A floating-ring, controlled-gap seal capable of sustaining the high differential pressure was selected for the turbine seal. Two overboard drain lines, on either side of the intermediate seal, are provided to dispose of the leakage past the primary and turbine seals. Gaseous hydrogen leakage through the turbine seal to overboard was estimated at 0.0065 lb/sec. To ensure a complete separation of the pump and turbine fluids, an intermediate seal consisting of two floating-ring seals was included. A gaseous helium purge is introduced into the intermediate seal to effect a high-pressure barrier between the two drain cavities.

Rotordynamics. An analysis of the dynamic characteristics of the rotor was conducted. The critical speeds were calculated with a finite element method. The shaft was approximated as a series of concentrated masses and inertias connected by elastic beam elements. Forward synchronous precession was assumed and the bearings were modeled as linear springs to ground. The gyroscopic effect of each rotating mass was included.

The critical speed locations and trends are given in Fig. 98. The curves show that the rotor operates between its first and second critical speed.

As a result of the 5:1 throttling requirement of the engine, the turbopump must be capable of operating steady state anywhere between the nominal design speed of 22,000 rpm and the minimum throttling speed of 10,850 rpm. A 20-percent "pad" was applied at both ends of the above range as a safety margin. Figure 98 shows that to keep the lower range free of critical speeds, the turbine bearing spring rate (K2) must be less than 90,000 lb/in. To provide a spring rate that low, the turbine bearing was "soft-mounted," i.e., in a flexible support. Although the second critical is above the 22,000-rpm nominal speed, to provide the desired 20-percent pad, the pump end bearing structure will be designed to ensure that its spring rate is above 200,000 lb/in.

Stress. The LOX turbopump was sized to meet the structural criteria noted in Table 43. The minimum thicknesses of the critical rotating parts, i.e., the

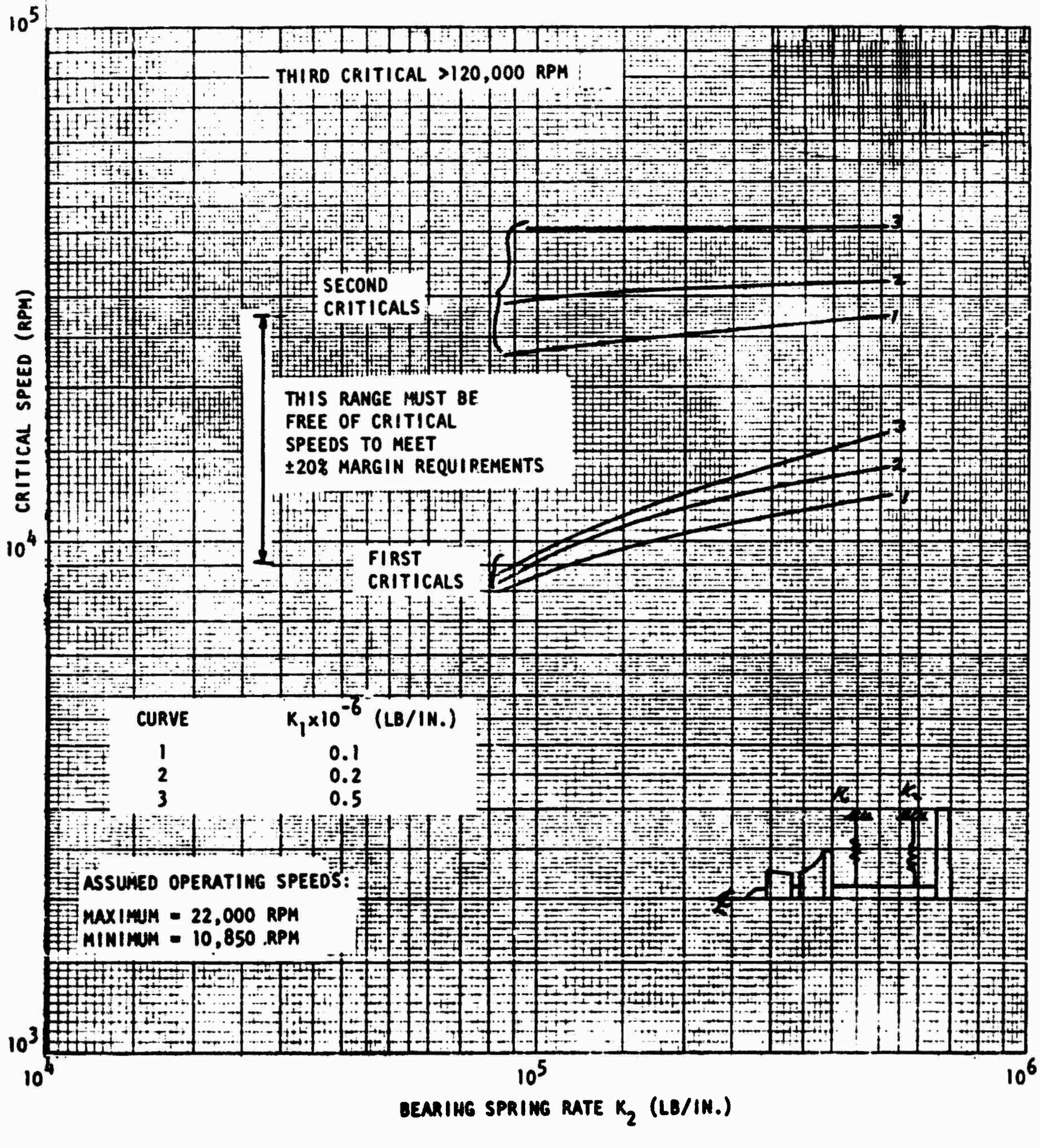


Figure 98. Double-Panel LOX Pump Critical Speeds

impeller and turbine disk, were established by mechanical constraints or vibration considerations. Thus, their burst speeds are significantly above the expected operating range (e.g., impeller burst speed is 90,000 rpm).

In selecting materials for turbine components, consideration was given to hydrogen environment embrittlement. The manifold material, Haynes 188, and the turbine disk material, A-286, both are noted for their resistance to hydrogen embrittlement.

Weight. A calculation of the oxidizer turbopump weight was made on the basis of the cross sections shown in Fig. 95. A breakdown of the estimated weights is included in Table 45.

TABLE 45. DOUBLE-PANEL LOX TURBOPUMP WEIGHTS

Inducer	0.4
Impeller	2.8
Shaft	1.6
Wheel	6.3
Misc. Parts	1.2
Total Rotating Parts	12.3
Volute	10.8
Bearing Carrier	17.5
Turbine Manifold	7.4
Total Static Parts	35.7
Total LOX Turbopump	48.0

Maintainability. Inspection of the LOX turbopump will be performed after 2 hours of operation or 60 starts. The turbopump design provides a visual inspection capability of both bearings through the speed transducer port using fiber optic techniques. The turbine rotor blades and nozzle vanes will be examined

through instrumentation ports by similar methods. The integrity of the shaft seals will be verified by gas leak checks.

After 10 hours of operation or 300 starts, the turbopump will be completely disassembled and the shaft seals, bearings, and impeller seals will be replaced. All other parts will be examined visually; rotating parts and pressure vessels will be dye-penetrant inspected to guard against incipient cracks or other signs of degradation.

SINGLE-PANEL ENGINE TURBOPUMPS

A design study was conducted to define conceptually the liquid hydrogen and liquid oxygen turbopumps for the AMPS 25,000-pound-thrust, 750-psia chamber pressure engine. Suction performance considerations and the objective of minimum size established the speed of the fuel turbopump at 75,000 rpm and the speed of the oxidizer turbopump at 22,000 rpm. The critical performance requirements of the turbopumps as dictated by engine considerations are presented in Table 46.

The general design philosophy was to use existing technology. In addition to the performance requirements presented in Table 46, the following criteria were applicable:

Time Between Inspections	2 hours, 60 starts
Time Between Overhaul	10 hours, 300 starts
Total Life	50 hours, 1500 starts
Inspection	5 percent of original cost
Overhaul	25 percent of original cost

The basic configurations of the turbopumps were established on the basis of the tradeoff studies conducted for the 1000-psia chamber pressure AMPS engine as discussed in Appendix C. Those studies had resulted in the selection of a three-stage centrifugal pump with a single-stage, axial-flow impulse turbine for the fuel turbopump, and a single-stage centrifugal pump with a single-row, partial-admission axial turbine for the oxidizer turbopump. Because the application and desired characteristics of the turbopumps for the 750-psia engine are similar, the above features were adopted for its turbomachinery, except the number of fuel pump stages was decreased to two, in response to the lower discharge pressure requirement.

TABLE 46. SINGLE-PANEL AEROSPIKE 750-PSIA P_c, LO_2/LH_2 ENGINE TURBOPUMP PERFORMANCE REQUIREMENTS

o **ENGINE INFORMATION**		
Type	Aerospike	
Thrust	25000 lb	
Chamber Pressure	750 psia	
Nozzle Area Ratio	110:1	
Engine Mixture Ratio	5.5	
Secondary Mixture Ratio	0.0	
Turbine Drive Cycle	Expander Topping Cycle	
Turbine Arrangement	Parallel	
o **PUMP REQUIREMENTS**	FUEL	OXIDIZER
Fluid	LH_2	LO_2
Flowrate, lb/sec	8.4	46.2
Inlet Pressure, psia	15	25
Discharge Pressure, psia	1575	1057
NPSH, ft	60	16
o **TURBINE REQUIREMENTS**		
Fluid	GH_2	GH_2
Inlet Temperature, °R	840	840
Inlet Pressure, psia	1156	1116
Exhaust Pressure, psia	890	900
o **THROTTLING REQUIREMENT**	5:1	

LH_2 Turbopump Design

Fluid Dynamic Design. The performance parameters and control dimensions of the LH_2 pump are presented in Table 47; in Fig. 99, the pump head coefficient and efficiency are given as a function of flow coefficient. The expressions for the coordinates are normalized relative to design point values to permit the application of the curves directly in the engine balance computer program.

The operating speed of the liquid hydrogen pump (75,000 rpm) was established at the maximum attainable with the available NPSH. The speed selection was based on a suction specific speed of 99,000 which corresponds to an inlet total head of one velocity head ($C_m^2/2g$). The technology base for this criterion has been established on the J-2 and J2S fuel pumps which have demonstrated satisfactory operation with one velocity head (NAS8-19).

With the two-stage pump, the specific speed per stage obtained was 1105, resulting in a pump overall efficiency of 65.7 percent. The head requirements were met with two 3.90-inch-diameter impellers, with a tip speed of 1280 ft/sec.

To meet the power requirement of the pump (1150 hp) at 75,000 rpm, a single-stage impulse turbine was designed with a wheel pitch diameter of 5.0 inches. The performance parameters and significant dimensions of the turbine are noted in Table 48 . The wheel tip speed/gas spouting velocity ratio (U/C_o) obtained was 0.503, resulting in a turbine efficiency of 78.6 percent. A turbine propellant flowrate of 4.9 lb/sec is required to meet the nominal pump power requirements. The efficiency of the fuel turbine as a function of U/C_o ratio is presented in Fig. 100. In Fig. 101, turbine performance is defined by plotting a torque parameter (ordinate) versus a speed parameter (along the abscissa).

Configuration Description. In Fig. 102, the final layout of the LH_2 turbopump is presented which defines the internal detail configuration of the selected design.

The pumping elements consist of an axial-flow inducer required to meet the NPSH requirements, followed by two centrifugal impellers. The hydrodynamic passages of the two impellers are identical, which allows machining them from identical castings into the various finished configurations, effecting tooling and production

TABLE 47. LH_2 TURBOPUMP PUMP PERFORMANCE PARAMETERS AND CRITICAL DIMENSIONS

Number of Stages	2
Rotating Speed, rpm	75,000
Flowrate, lb/sec	8.4
Flowrate, gpm	856
Isentropic Head, ft.	48,600
Discharge Pressure, psia	1591
Inlet Pressure, psia	25
Impeller Tip Diameter, in.	3.90
Impeller Tip Speed, fps	1280
Specific Speed per Stage	1105
Stage Efficiency, percent	67.5
Overall Efficiency, percent	65.7
Horsepower	1150
Minimum NPSH required, ft.	60
Inlet Velocity Head, ft.	53.5
Design Head Coefficient	0.50
Design Flow Coefficient	0.120

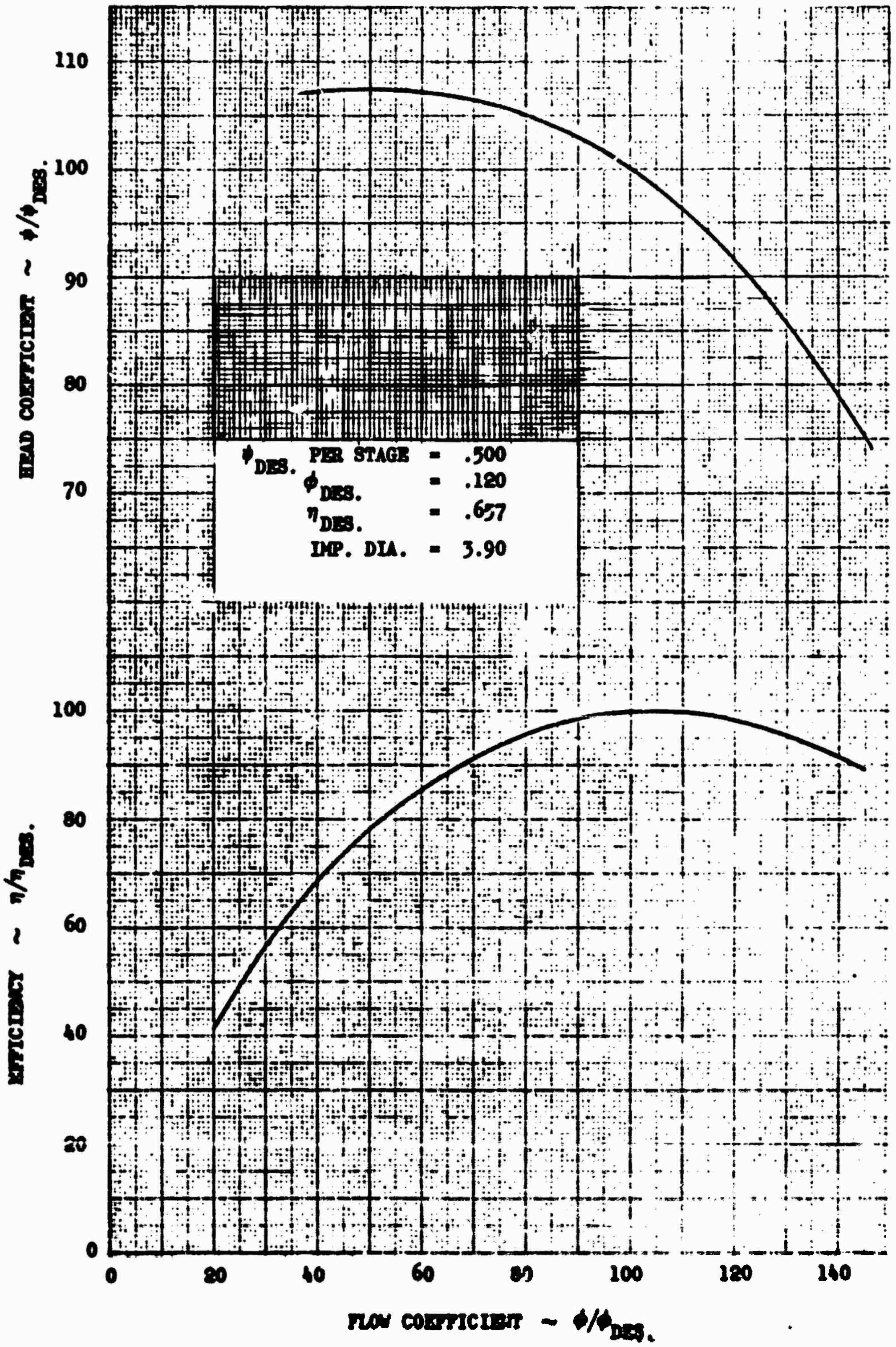


Figure 99. Single-Panel LH_2 Pump (75,000 rpm, two stages)

TABLE 48. SINGLE-PANEL LH_2 TURBOPUMP TURBINE PERFORMANCE PARAMETERS AND CRITICAL DIMENSIONS

Inlet Temperature, R	840
Inlet Pressure, psia	1156
Exhaust Temperature, R	790
Exhaust Pressure, psia	890
Pressure Ratio	1.3
Speed, rpm	75,000
Blade Speed, fps	1637
Velocity Ratio	0.503
Efficiency, percent	78.6
Flowrate, lb/sec	4.9
Horsepower	1150
Admission, percent	100
Stress, Factor, $AAN^2 \times 10^{-9}$	55.0
Nozzle Chord, in.	0.350
Nozzle Height, in.	0.292
Rotor Blade Chord, in.	0.350
Rotor Blade Height, in.	0.626
Manifold Tours Dia., in.	0.900
Manifold Inlet Dia., in.	1.500

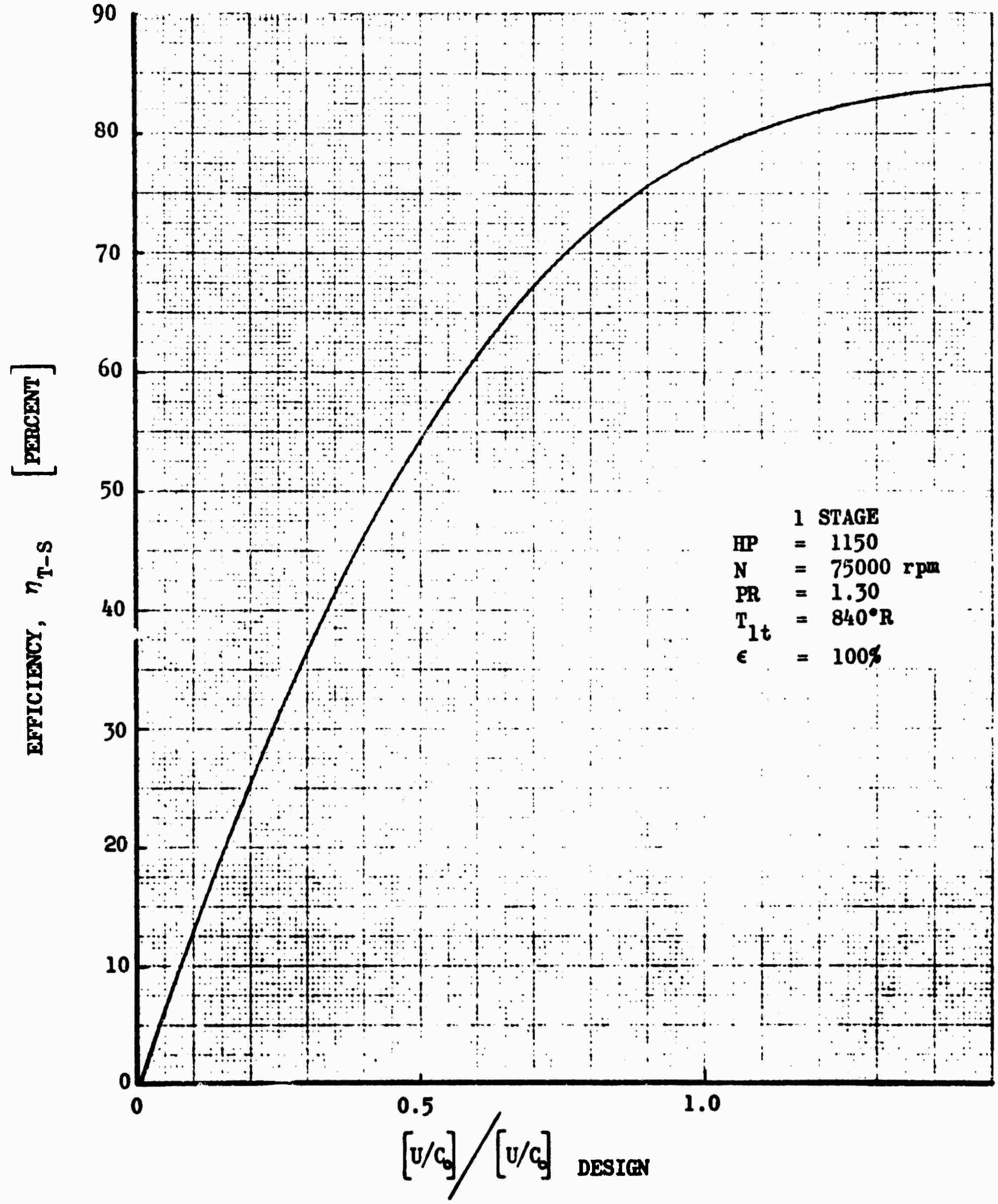


Figure 100. Single-Panel Fuel Turbine Efficiency

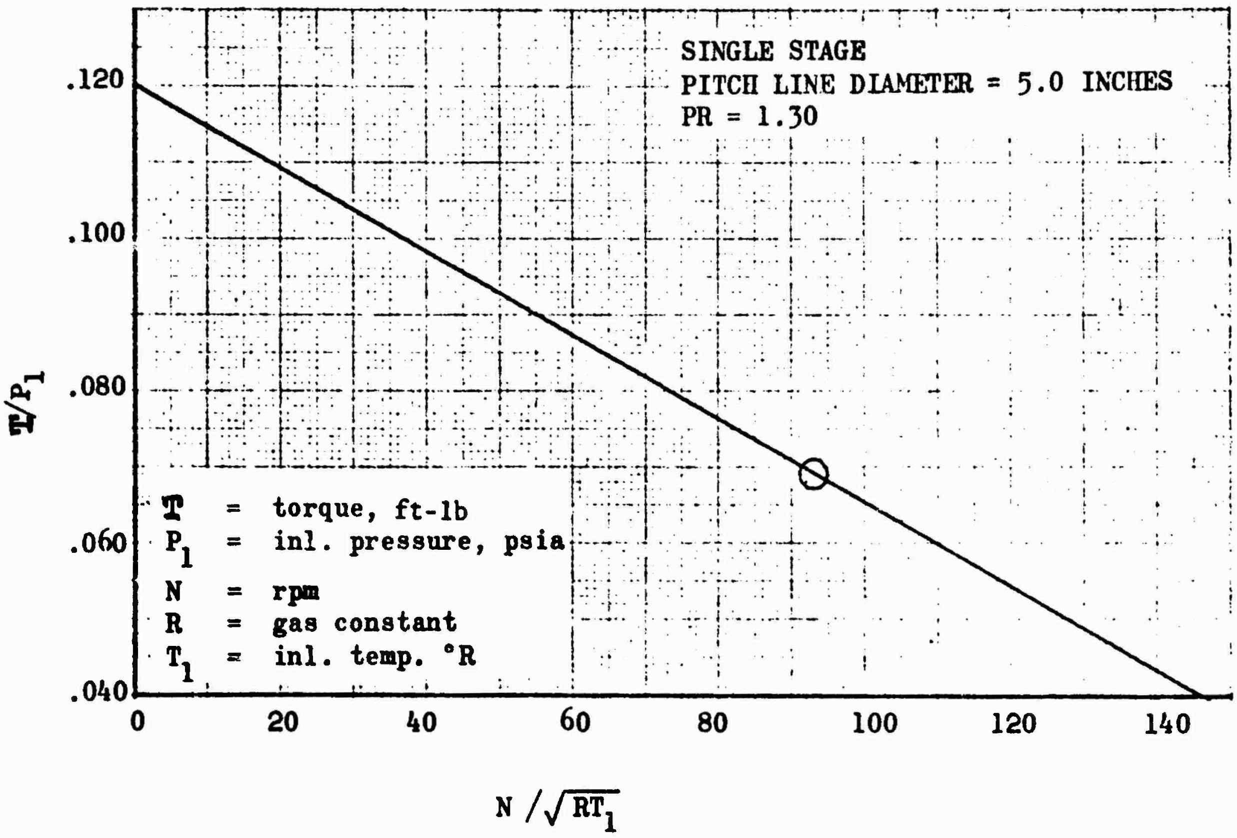


Figure 101. Single-Panel Fuel Turbine Performance

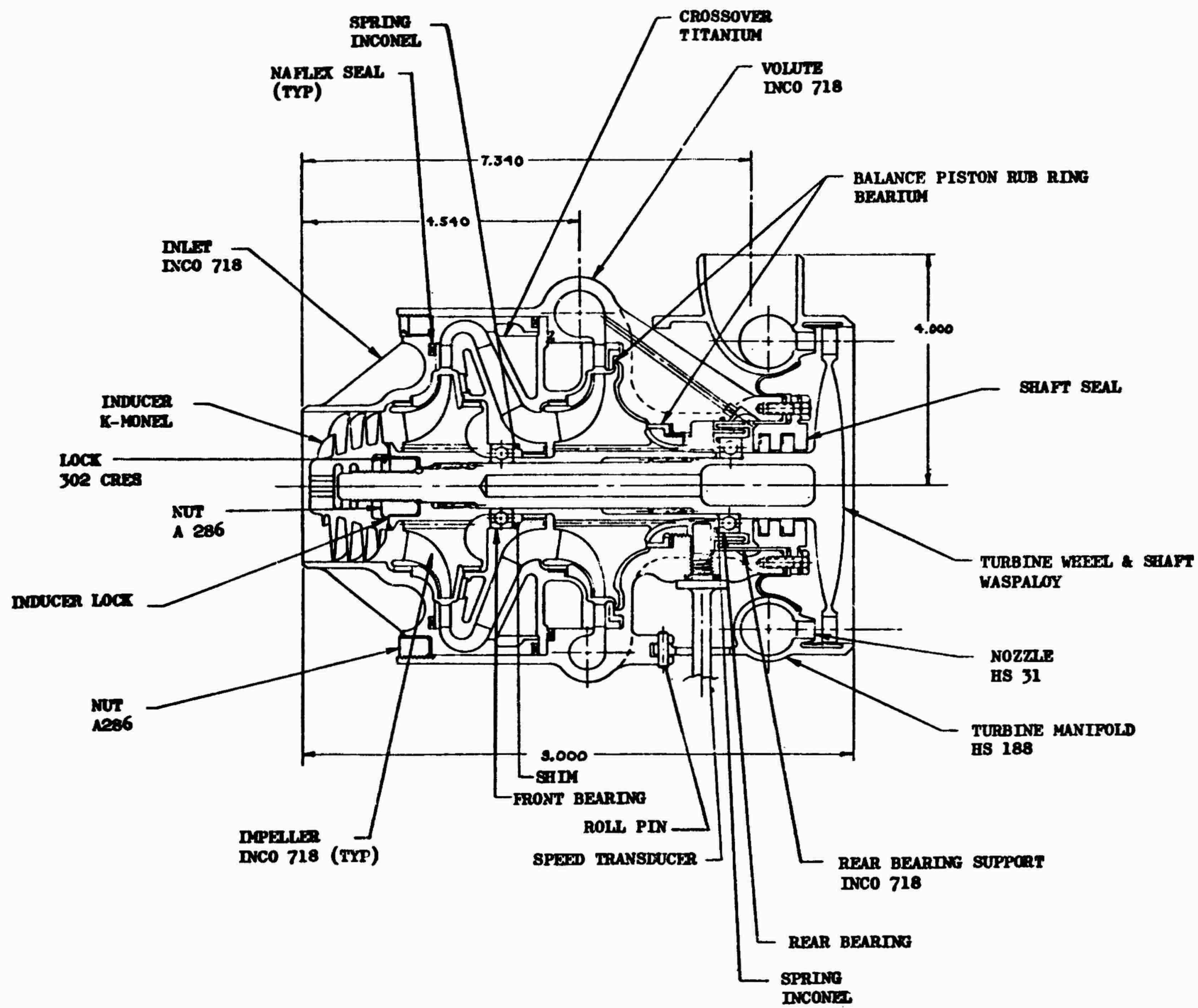


Figure 102. Fuel Turbopump, Single-Panel Engine

cost savings. The first-stage impeller is followed by a crossover section, consisting of radial diffuser vanes followed by a vaneless turning passage and a radially inward-flow, vaned diffuser. The second diffuser row in the crossover introduces the same amount of whirl into the liquid as the inducer, to permit the use of identical impellers. The second impeller discharges into a radial-vaned diffuser from where the fluid is collected in a scroll-shaped volute and delivered through a single discharge pipe.

The crossover also serves as the support for the front bearing, and can be cast from INCO 718 or welded from either Inconel or titanium. For the purposes of this conceptual study, a welded titanium construction was assumed. Also within current state of the art is the casting of these parts from titanium, but some casting development effort would be required. This possibility should be reviewed at the time of the final detail design in the context of specific program objectives and schedules and in light of further advances in titanium casting technology.

The impeller tip speed required to generate the desired head was 1280 ft/sec, sufficiently low to permit using conventional INCO 718 castings for the impellers. The inducer is machined from a pancake forging. Of the several materials which could be used, K-monel offered the best compromise of high strength and machinability and was therefore selected. Both impellers as well as the two bearing inner races, are axially preloaded against the shaft shoulder with a nut located forward of the first-stage impeller. Splines are used to transmit the torque from the shaft to the impellers. The inducer is threaded directly on the shaft and locked with a deformed lock-tab located between the inducer and the first-stage impeller. The thread direction is such that the inertia effect during start as well as the operational torque loads tend to tighten the inducer. This concept of retaining the inducer has been successfully demonstrated on Rocketdyne's Mark 36 liquid fluorine pump at speeds up to 104,000 rpm (NAS3-12022).

The volute, which also serves as the rear bearing support and shaft seal housing, is cast from INCO 718, as is the inlet to the pump. These two parts are in-place welded to the engine inlet and discharge ducts, respectively, which act as mounting points for the turbopump.

Shaft axial thrust control is affected by locating the wear ring behind the first-stage impellers at the proper diameters. A self-compensating balance piston is incorporated into the second-stage impeller to absorb any residual axial loads. To operate the balance piston, high-pressure liquid hydrogen is bled from the discharge of the second-stage impeller and passed through an orifice at the outer diameter of the impeller into the balance piston cavity. From the balance piston cavity, the fluid is routed through a low-pressure orifice and returned into the eye of the impeller through drilled holes in the impeller hub. If there is an unbalanced axial load toward the pump inlet, the rotor moves forward, closing the high-pressure orifice gap and opening the low-pressure orifice gap. The resulting decrease in the balance piston cavity pressure level introduces a correcting axial force toward the turbine end. Conversely, if there is an unbalanced axial load toward the turbine, the rotor moves aft, opening the high-pressure orifice gap and closing the low-pressure orifice gap. As a result, the cavity pressure increases until the unbalance load is cancelled.

The turbine manifold is mounted on the pump volute through a radial pin joint which allows thermal differential expansion. It is fabricated by welding preformed details of Haynes 188 material. The nozzle is cast as an integral piece of Haynes 31 and is electron-beam welded into the manifold. The turbine disk and the shaft are rough machined from Waspaloy forgings and welded together in the vicinity of the shaft rider seal. The turbine rotor blades are machined integral with the disk.

Mounting of the turbopump to the engine is accomplished by in-place welding rigid engine ducting to the pump inlet and turbine inlet.

Bearings. The pump rotor is supported on two angular contact ball bearings. The bearings selected for this pump are identical to those selected for the double-panel engine hydrogen pump. The primary objective was to fulfill the requirements for high-speed operation and long life.

Detailed dimensions of these bearings, materials, and fatigue life predictions are presented in the double-panel engine hydrogen pump design description (page 208).

Shaft Seals. As in the design of the double-panel engine hydrogen pump, a double-floating, controlled-gap seal was incorporated in this pump design. Supporting reasons for this selection are presented in the design description of the double-panel engine hydrogen pump (page 214).

Rotordynamics. An analysis of the dynamic characteristics of the rotor was conducted. The critical speeds were calculated with a finite element method. The shaft was approximated as a series of concentrated masses and inertias connected by elastic beam elements. Forward synchronous precession was assumed and the bearings were modeled as linear springs to ground. The gyroscopic effect of each rotating mass was included.

The critical speed locations and trends are given in Fig. 103. The curves show that the rotor will operate between its second and third critical speed.

As a result of the 5:1 throttling requirement of the engine, the turbopump must be capable of operating steady state anywhere between the nominal design speed of 75,000 rpm and the minimum throttling speed of 36,300 rpm.

A 20 percent "pad" was applied at both ends of the above range as a safety margin. Figure 103 shows that to keep the lower range free of critical speeds. the turbine end bearing spring rate (K_2) must be approximately 50,000 lb/in. or smaller, and the pump bearing spring rate (K_1) must be 100,000 lb/in. or smaller. The latter criterion was satisfied without special design features. The turbine end bearing was soft mounted to ensure a spring rate below 50,000 lb/in.

Stress. The pump shaft and turbine wheel are integral and made of Waspaloy material. The coefficient of thermal contraction of the Waspaloy shaft matches that of the INCO 718 impellers, thereby minimizing piloting and axial stack problems. The shaft factor of safety was determined utilizing the modified Soderberg equation which is based on the maximum shear failure theory. The minimum shaft factor of safety was 2.0 compared with a minimum requirement of 1.4. The limitation on maximum speed was imposed by the fourth diametral mode critical frequency of the turbine disk, which was calculated at 90,900 rpm. A 15 percent margin is desirable relative to the disk material frequency; as a result, the maximum allowable steady-state operating speed was limited to 77,200 rpm.

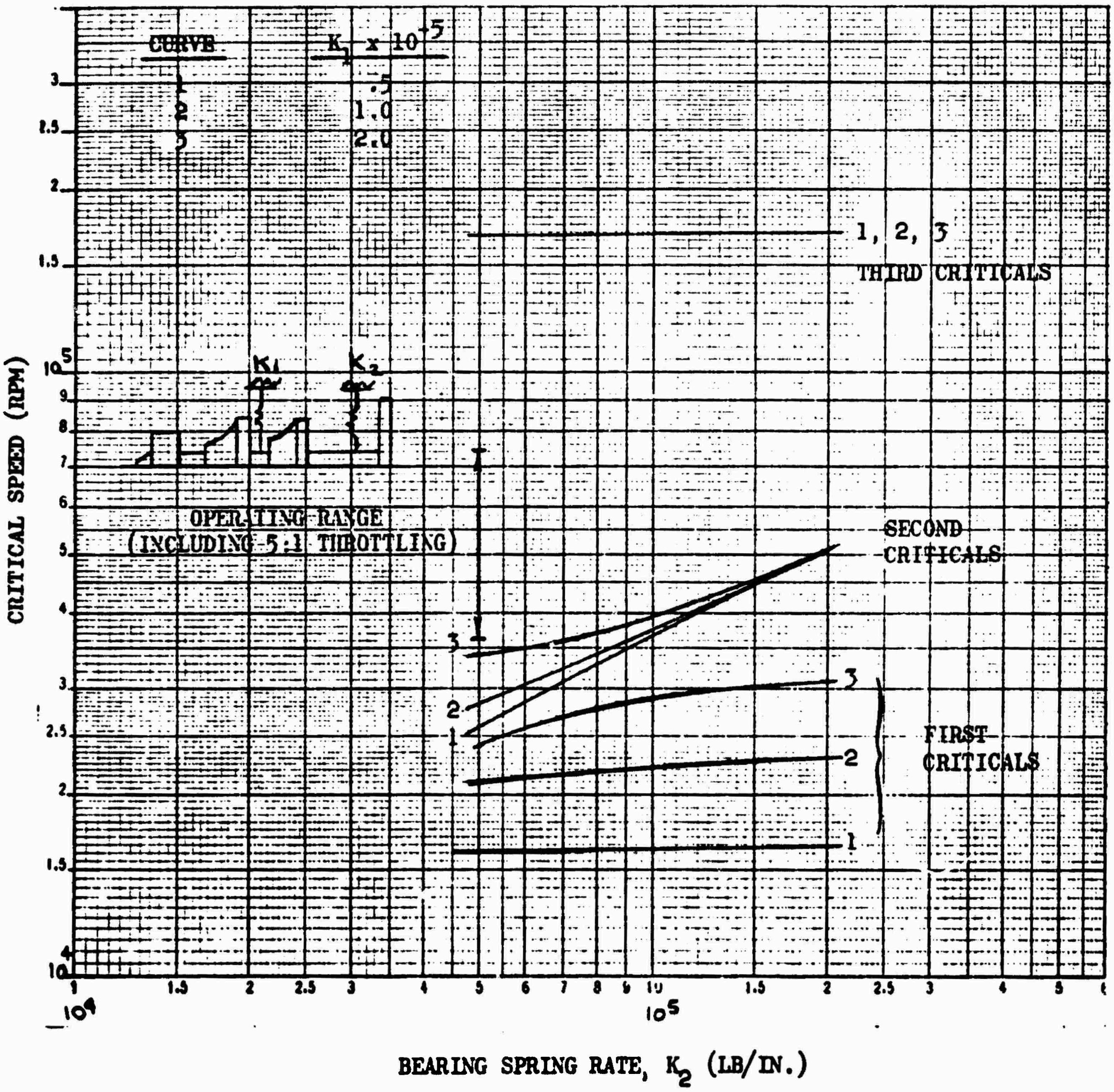


Figure 103. Single-Panel Engine Fuel Turbopump Rotor Critical Speeds

The turbine manifold is fabricated from Haynes 188 material to afford maximum protection against the hydrogen environment embrittlement. The nozzle is cast Haynes 31 material.

Weight. The weight of the fuel turbopump was calculated on the basis of the cross sections shown in the layout of Fig. 102. The total weight of the turbopump was estimated at 32 pounds.

Maintainability. Inspection and overhaul procedures and schedule for this pump are identical with those discussed for the double panel engine hydrogen pump (page 217).

LOX Turbopump Design

Fluid Dynamic Design. The performance parameters and control dimensions of the LOX turbopump are presented in Table 49. In Fig. 104 the pump head coefficient and efficiency are given as a function of flow coefficient. The expressions for the coordinates are normalized relative to design point values to permit the application of the curves directly in the engine balance computer program.

The nominal operating speed of the liquid oxygen pump (22,000 rpm) was established at the maximum attainable with the available NPSH. The speed selection was based on a suction specific speed of 46,000 which corresponds to an inlet condition of 2.5 velocity heads ($C_m^2/2g$).

The design specific speed is 1230, resulting in a pump efficiency of 68 percent. The head requirements were met with a 3.81-inch-diameter impeller, with a tip speed of 366 ft/sec.

To meet the power requirement of the pump (256 hp) at 22,000 rpm, a partial-admission, single-stage impulse turbine was designed with a wheel pitch diameter of 8.00 inches. The performance parameters and significant dimensions of the

TABLE 49. SINGLE-PANEL LOX TURBOPUMP PERFORMANCE PARAMETERS AND CRITICAL DIMENSIONS

PUMP:	
Number of Stages	1
Rotating Speed, rpm	22000
Flowrate, lb/sec	46.2
Flowrate, gpm	291
Isentropic Head, ft.	2080
Inlet Pressure, psia	25
Discharge Pressure, psia	1062
Impeller Tip Diameter, in.	3.81
Impeller Tip Speed, fps	366
Specific Speed Per Stage	1230
Stage Efficiency, percent	68
Overall Efficiency, percent	68
Horsepower	256
Minimum NPSH Required, ft.	16
Inlet Velocity Head, ft.	5.24
Design Head Coefficient	0.50
Design Flow Coefficient	0.12

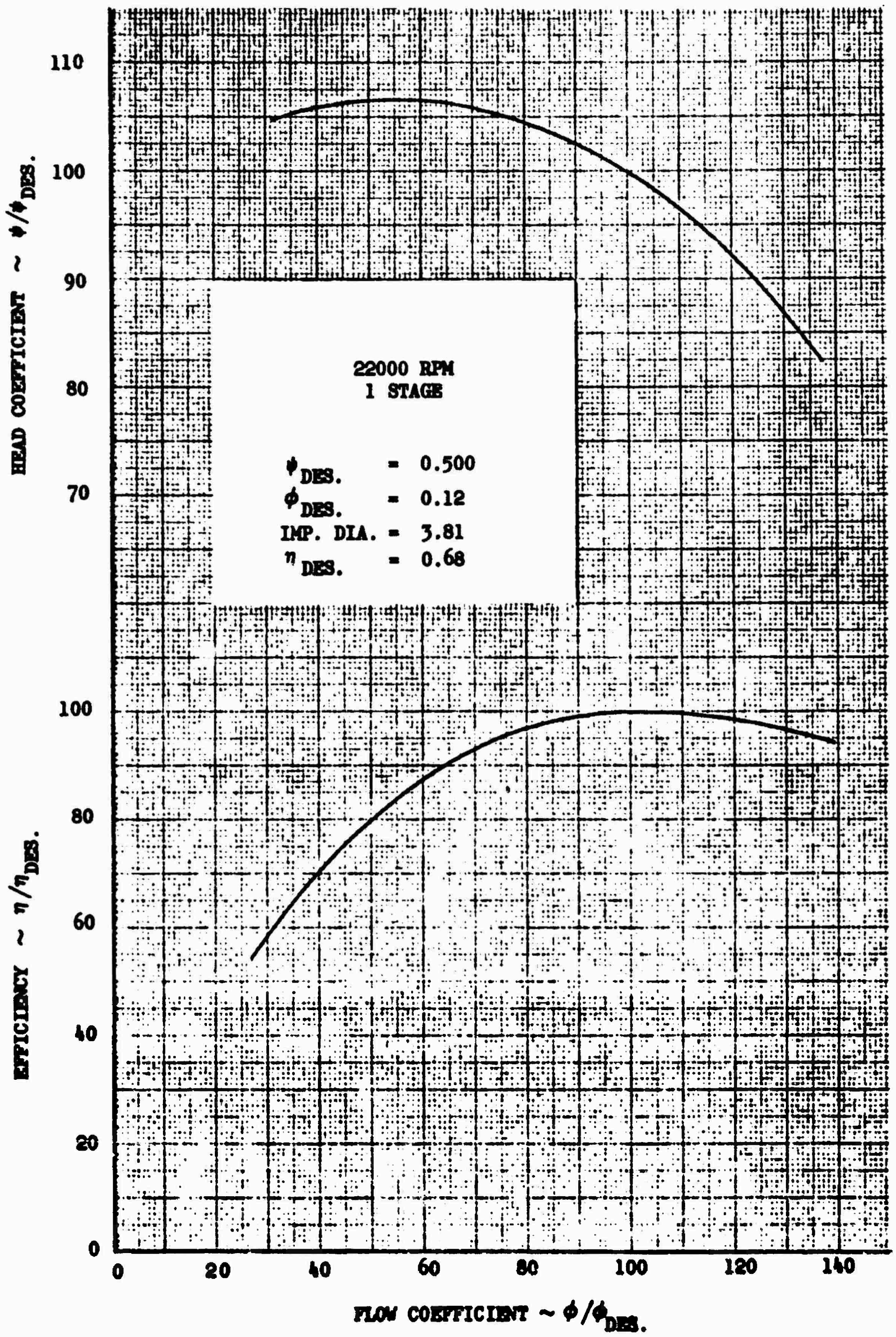


Figure 104. Single-Panel LOX Pump Performance

turbine are presented in Table 50. The wheel tip speed-to-gas spouting velocity ratio (U/C_o) obtained was 0.262, resulting in a predicted turbine efficiency of 59.7 percent. A turbine propellant flowrate of 1.77 lb/sec is required to meet the nominal pump power requirements. The efficiency of the fuel turbine as a function of U/C_o ratio is presented in Fig. 105. In Fig. 106, turbine performance is defined by plotting a torque parameter (ordinate) versus a speed parameter (along the abscissa).

Configuration Description. In Fig. 107, the final layout of the LOX turbopump is presented which defines the internal detail configuration of the selected design.

The pumping elements consist of an axial flow inducer required to meet the NPSH requirements, followed by a centrifugal impeller. Fluid from the impeller is discharged into a vaned radial diffuser, collected in a scroll-shaped volute, and delivered through a single discharge pipe.

The inducer is machined from K-monel forging which offers an excellent combination of strength and resistance to chemical reaction in LOX. The inducer is threaded directly on the shaft and secured with a sheet metal lock-tab located between the inducer and impeller. The thread direction is such that the inertia effect during start as well as the operational torque loads will tend to tighten the inducer.

The impellers, volute, and the diffuser/bearing carrier subassembly are cast from INCO 718. To facilitate casting and to obtain better tolerance control, the diffuser is cast separately and joined to the bearing carrier by welding. The volute is in-place welded to the engine inlet and discharge ducts. These joints act also as the mounting points for the turbopump.

Shaft axial thrust control was effected by locating the impeller rear wear ring diameter so that the net axial thrust becomes zero. Any residual thrust due to calculation error and off-design operation is carried by the front bearing. Kel-F is used for both impeller wear rings, permitting close wear ring clearances

TABLE 50. SINGLE-PANEL LOX TURBOPUMP TURBINE PERFORMANCE PARAMETERS AND CRITICAL DIMENSIONS

Inlet Temperature, R	840
Inlet Pressure, psia	1116
Exhaust Temperature, R	808
Exhaust Pressure, psia	900
Pressure Ratio	1.24
Speed, rpm	22000
Blade Speed, fps	768
Velocity Ratio	0.262
Efficiency, percent	59.7
Flowrate, lb/sec	1.77
Horsepower	256
Admission, percent	20
Stress Factor, $AAN^2 \times 10^{-9}$	6.49
Nozzle Chord, in.	0.350
Nozzle Height, in.	0.367
Rotor Blade Chord, in.	0.300
Rotor Blade Height, in.	0.534
Manifold Torus Dia., in.	0.650
Manifold Inlet Dia., in.	0.900

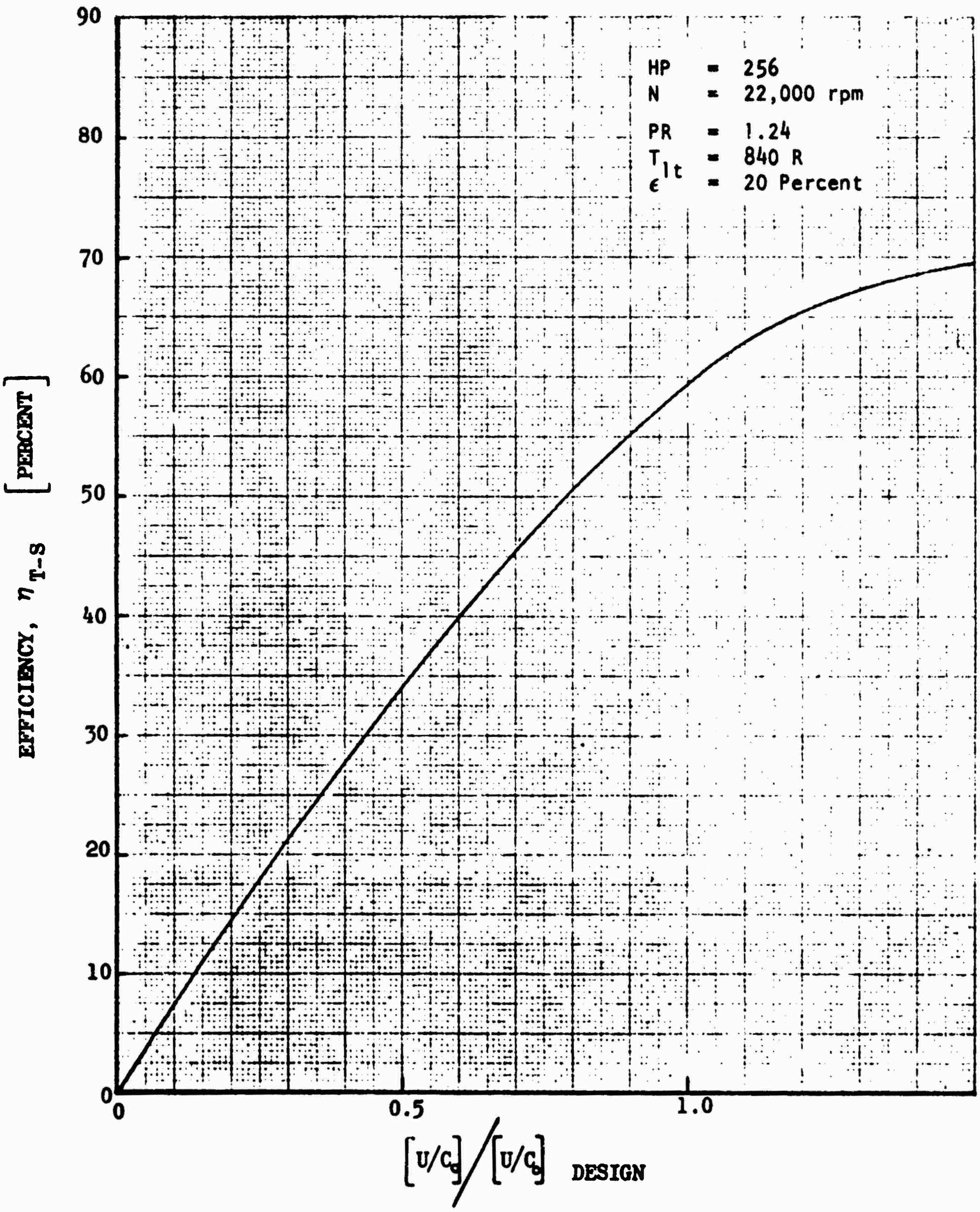


Figure 105. Single-Panel Oxidizer Turbine Efficiency

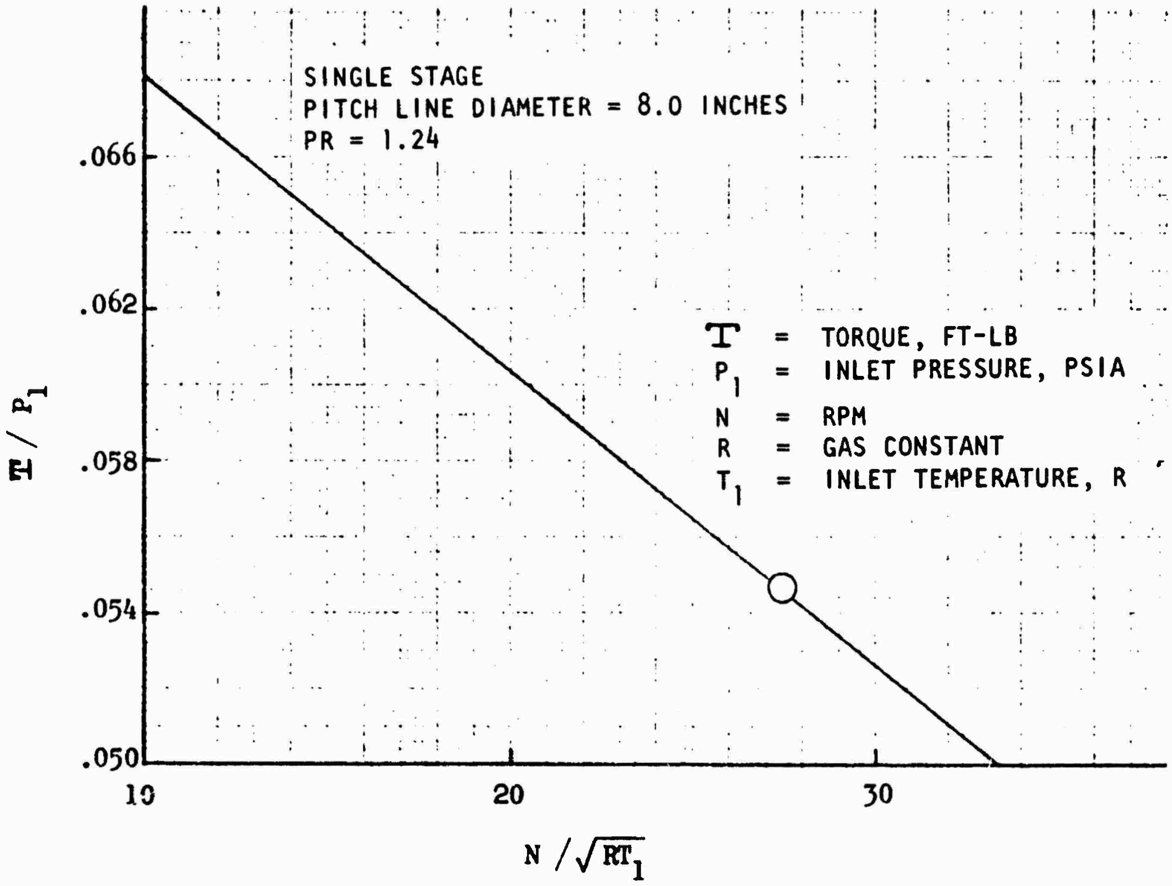


Figure 106. Single-Panel LOX Turbine Performance

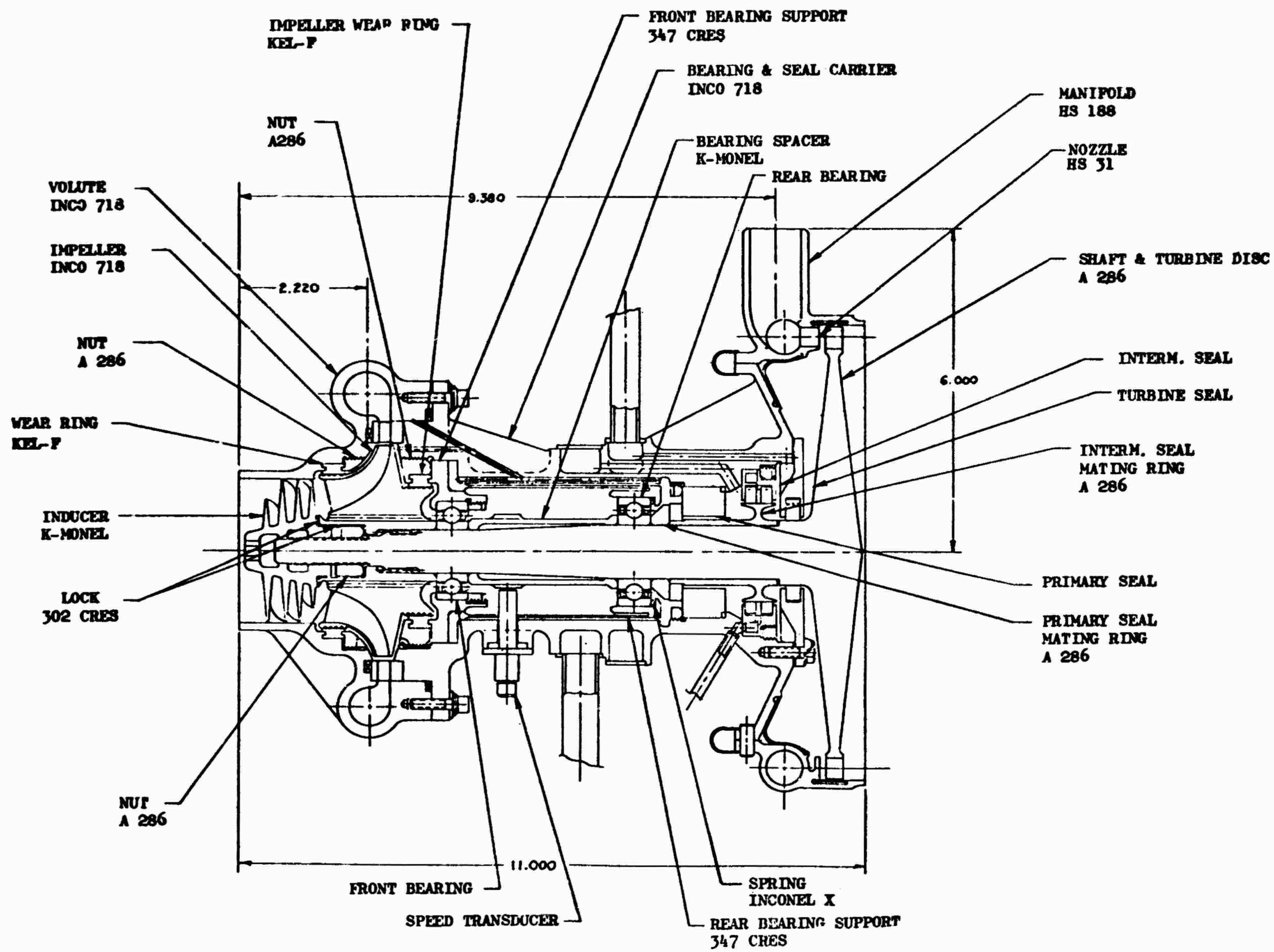


Figure 107. LOX Turbopump, Single-Panel Engine

which are desirable from a performance standpoint, without the hazards presented by metal-to-metal contact.

Rotordynamic considerations dictated a low radial spring constant for the turbine end bearing. To comply with this requirement, the bearing was "soft-mounted," i.e., the bearing support is made of a flexible tubular cross section, cantilevered from the area of the front bearing. Excessive radial motion of the rear bearing was avoided by providing only a small clearance between the bearing support OD and the bearing carrier ID.

The turbine manifold is mounted on the bearing carrier flange through a pin joint which allows radial growth of the manifold without introducing appreciable thermal stresses. External leakage through the pin joint is prevented by a welded sheet metal seal on the inside of the joint. An external cover is welded on the joint to prevent moisture from collecting and freezing between the joint and the internal seal. The manifold is fabricated by welding preformed details of Haynes 188 material. The nozzle is cast from Haynes 31 and electron-beam welded into the manifold.

The turbine disk and the shaft are rough machined from A-286 forgings and joined by welding near the disk hub. The turbine rotor blades are machined integral with the disk.

Bearings. The pump rotor is supported on two Conrad-type ball bearings of 20-mm bore at the pump end and 25-mm bore on the turbine end. Detailed dimensions, materials selection, and fatigue life predictions of the bearings selected for this turbopump are identical to those presented in the design discussion for the double-panel engine oxygen turbopump (page 221).

Cage wear is another life-limiting factor for the bearings. Although LOX bearings have in the past operated successfully for 15 hours, the start requirements were limited to approximately 100. Therefore, bearing replacement is scheduled after 10 hours or 300 starts, even though the metal fatigue curves given in Fig. 96 and 97 indicate a substantial margin.

Shaft Seals. The pump and turbine fluids, LOX and GH_2 are not compatible chemically; therefore, they must be prevented from mixing. A face-type hydrodynamic pump seal, a purged intermediate seal, and floating-ring controlled gap turbine seal are used to provide absolute separation between the two areas. Background information pertaining to the selection of this seal package is presented in the double-panel engine oxygen pump description (page 228).

Rotordynamics. An analysis of the dynamic characteristics of the rotor was conducted. The critical speeds were calculated with a finite element method. The shaft was approximated as a series of concentrated masses and inertias connected by elastic beam elements. Forward synchronous precession was assumed and the bearings were modeled as linear springs to ground. The gyroscopic effect of each rotating mass was included.

The critical speed locations and trends are given in Fig. 108. The curves show that the rotor will operate between its first and second critical speed.

As a result of the 5:1 throttling requirement of the engine, the turbopump must be capable of operating steady state anywhere between the nominal design speed of 22,000 rpm and the minimum throttling speed of 9050 rpm. A 20-percent pad was applied at both ends of the above range as a safety margin. Figure 108 shows that, to keep the lower range free of critical speeds, the turbine bearing spring rate (K2) must be less than 50,000 lb/in. To make the spring rate that low, the turbine bearing was "soft-mounted", i.e., in a flexible support. Although the second critical is above the 22,000-rpm nominal speed, to provide the desired 20-percent pad, the pump end bearing structure will be designed to ensure that its spring rate is 150,000 lb/in. or higher.

Stress. The LOX turbopump was sized to meet the structural criteria noted in Table 41. The minimum thicknesses of the critical rotating parts, i.e., the impeller and turbine disk, were established by mechanical constraints or vibration considerations. Thus, their burst speeds are significantly above the expected operating range.

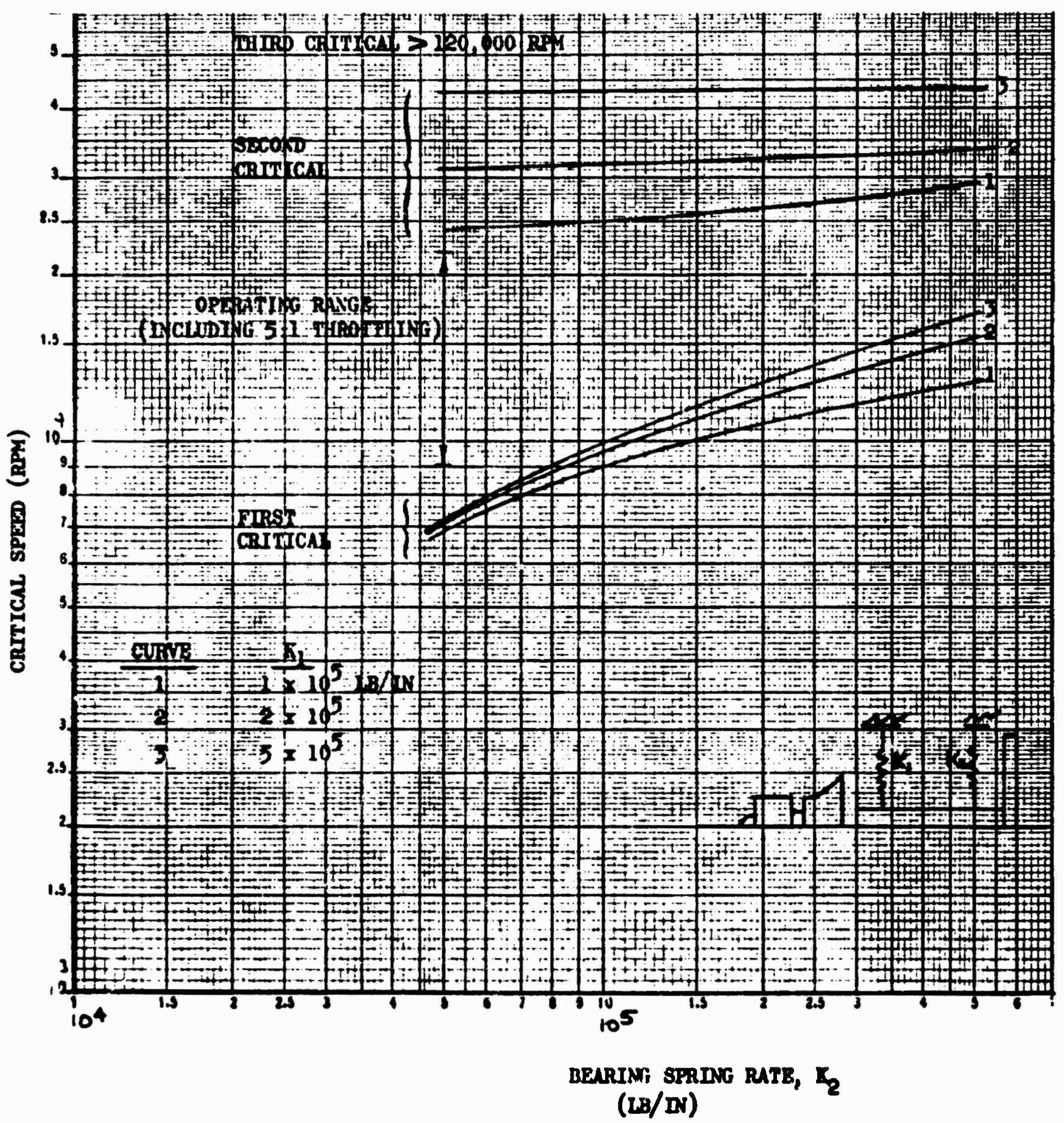


Figure 108. Single-Panel Engine LOX Turbopump Rotor Critical Speeds

In selecting materials for turbine components, consideration was given to hydrogen embrittlement. The manifold material, Haynes 188, and the turbine disk material, A-286, are both noted for their resistance to hydrogen embrittlement.

Weight. A calculation of the oxidizer turbopump weight was made on the basis of the cross-sections shown in Fig. 107. The total weight of the turbopump was estimated at 45 pounds.

Maintainability. Inspection and overhaul procedures and schedule for this pump are identical to those discussed for the double-panel engine oxygen pump (page 233).

CONTROLS DESIGN AND ANALYSIS

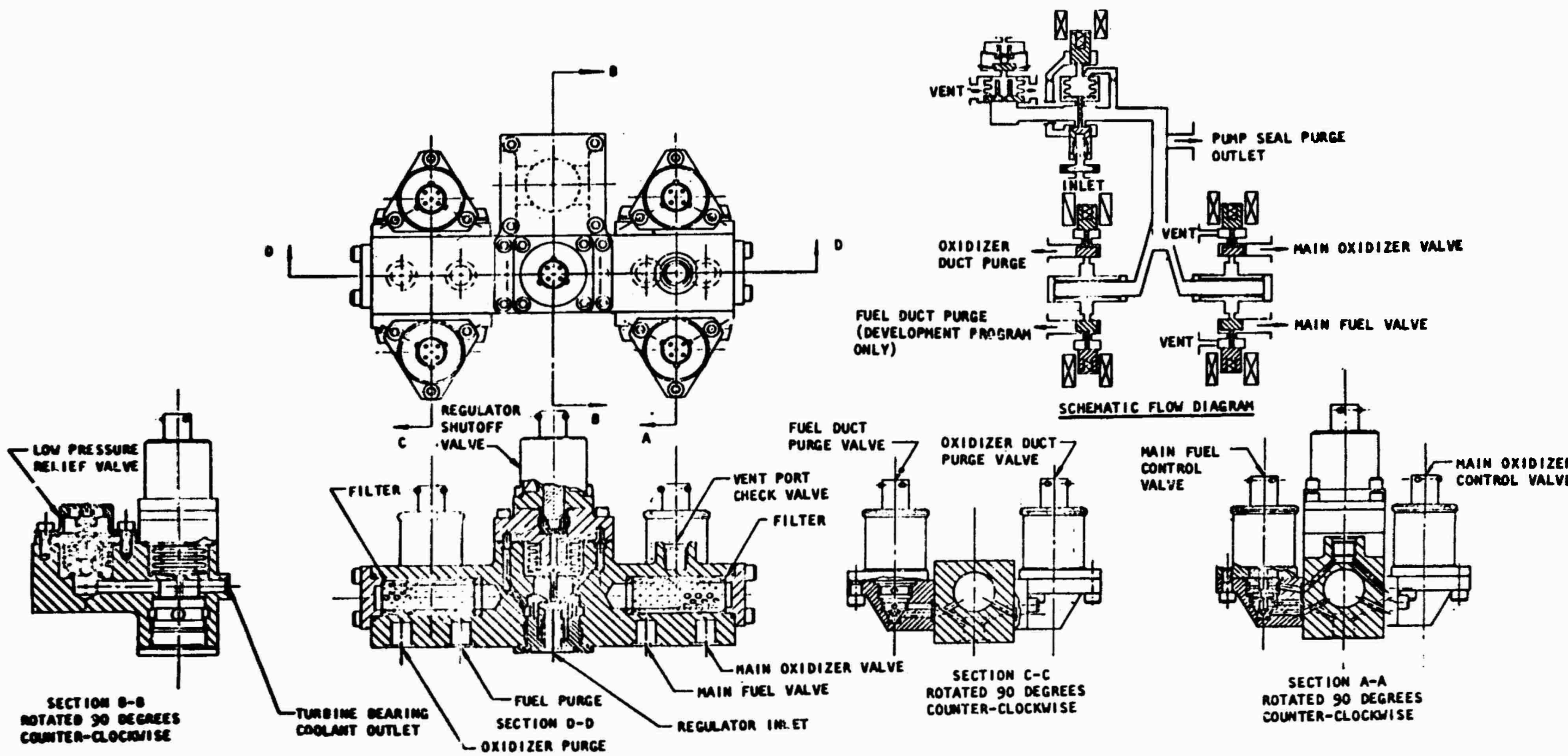
Preliminary design layouts and technical descriptions are presented for the following components:

1. Pneumatic Control Assembly
2. Oxidizer Turbine Inlet Control Valve
3. Turbine Bypass Control Valve
4. Main Fuel Valve
5. Main Oxidizer Valve

Identical engine system control components are required for both the double- and single-panel engine systems. Only minor dimensional variations are required in the oxidizer turbine inlet control valve and the turbine bypass valve to accommodate the differences in valve inlet pressure and pressure drop requirements.

Pneumatic Control Assembly

The pneumatic control assembly consists of a pressure regulator with solenoid on/off control; a low-pressure, high-capacity relief valve; two solenoid-operated, three-way valves to control the main engine propellant valves; two solenoid-operated, two-way valves to purge the propellant ducts; two filters to protect the solenoid valves; and one filter at the regulator inlet. A layout drawing is shown in Fig. 109.



2. FLUID PRESSURE:
OPERATING: 750 ±50 PSIA
PROOF: 1250 ±50 PSIA
1. FLUID HANDLED: HELIUM AT -260 TO +140 F

Figure 109. Pneumatic Control Assembly

The pneumatic pressure regulator is designed to regulate downstream pressure to 750 ±50 psig with flow demands from zero to 0.12 lb/sec, with an inlet pressure from 3600 to 900 psia.

The downstream pressure is controlled by a pressurized bellows which is supplied by inlet pressure and discharges to outlet pressure. The correct control pressure is obtained by orificing the flow into and out of the bellows cavity. The orifices are sized to provide a control pressure that compensates for the changes in supply pressure. The control pressure is opposing outlet or regulated pressure; thus, if outlet pressure decreases below 750 psig, the inlet flow area will increase to correct the pressure decrease and, inversely, if outlet pressure increases, the inlet flow area will decrease to reduce the outlet pressure.

The regulator is an all-metal construction. The materials selected are similar to those used on the J-2 regulator which is also designed for cryogenic service (NAS8-19). The J-2 regulator is fully qualified and has proved to be very successful on numerous flights. The regulator solenoid valve is a normally closed, two-way valve that opens upon electrical command. In the normally closed position gas is prevented from pressurizing the bellows, the main poppet remains closed, and the regulator serves as a positive shutoff. Regulator outlet pressure is zero Energizing the regulator solenoid valve pressurizes the bellows, the main poppet opens to pressurize the downstream pneumatic system, and the regulator functions as a regulating valve. The regulator and its solenoid valve are all-metal construction, except for electrical insulation, with flat-lapped poppets and seats for low leakage.

The low-pressure relief valve is located directly downstream of the pressure regulator. The relief valve is designed to crack or start to open at 825 psig and be fully open at 900 psig, with the flow area sufficient to expel the flow through a failed-open regulator with an upstream pressure of 3600 psia. The relief valve ensures that the downstream system pressure will not exceed 900 psig even if the regulator fails wide open.

The relief valve utilizes a pilot valve which will relieve the balance pressure within the bellows at the crack pressure, thus balancing the main poppet to open.

The three-way solenoid valves supply pressure to, and vent, the main propellant valve-actuating piston cavities--one for the fuel and one for the oxidizer. The valves have metal seats and poppets, flat-lapped to achieve low leakage.

The two-way solenoid valves supply flow to purge the propellant ducts during engine shutdown. These valves are identical to the three-way propellant valve control valves except the vent port is blocked off. Although a hydrogen-side purge is not planned in the flight application, the pneumatic control unit design layout drawing shows a hydrogen-side purge control valve for use during ground test development. This purge control valve can be easily removed for the flight operation phase when and if test experience demonstrates this as possible.

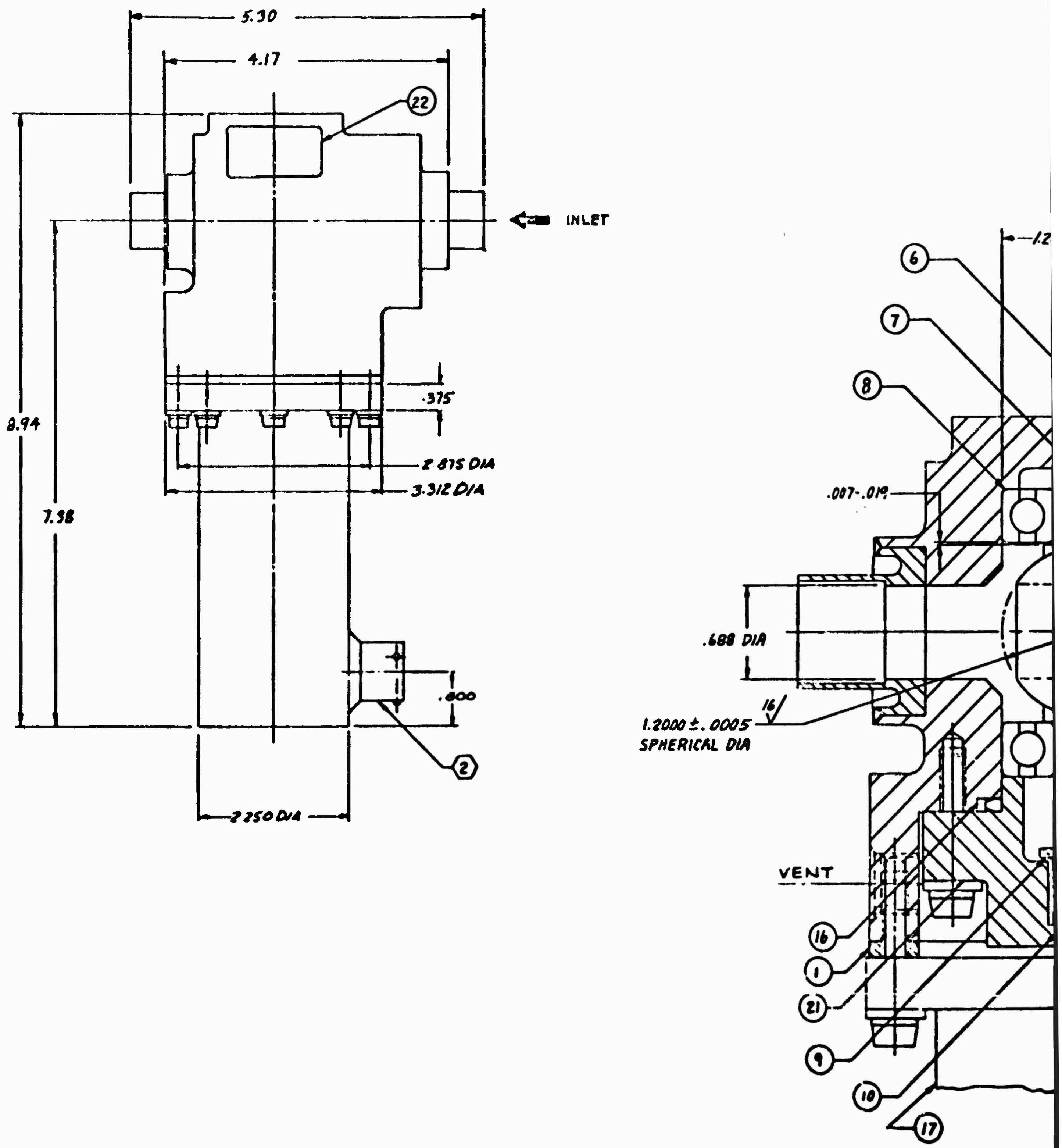
Turbine Bypass and Turbine Inlet Control Valves

These valves control the flow of hot gas to the oxidizer and fuel pump turbines and the turbine bypass flow. The designs are ball-type valves actuated by a-c motors through planetary gear systems. Position feedback potentiometers are provided for control and instrumentation. Design details of these valves are shown in Fig.110 and 111.

A ball-type throttle valve was selected for direct motor actuation and low pressure drop.

The design features an integral ball and shaft and is easily replaced for modification of throttling characteristics. Thrust and radial loads on the shaft produced by the flow of the hot gas are transmitted to the housing by means of ball bearings so that friction is kept to a minimum.

1



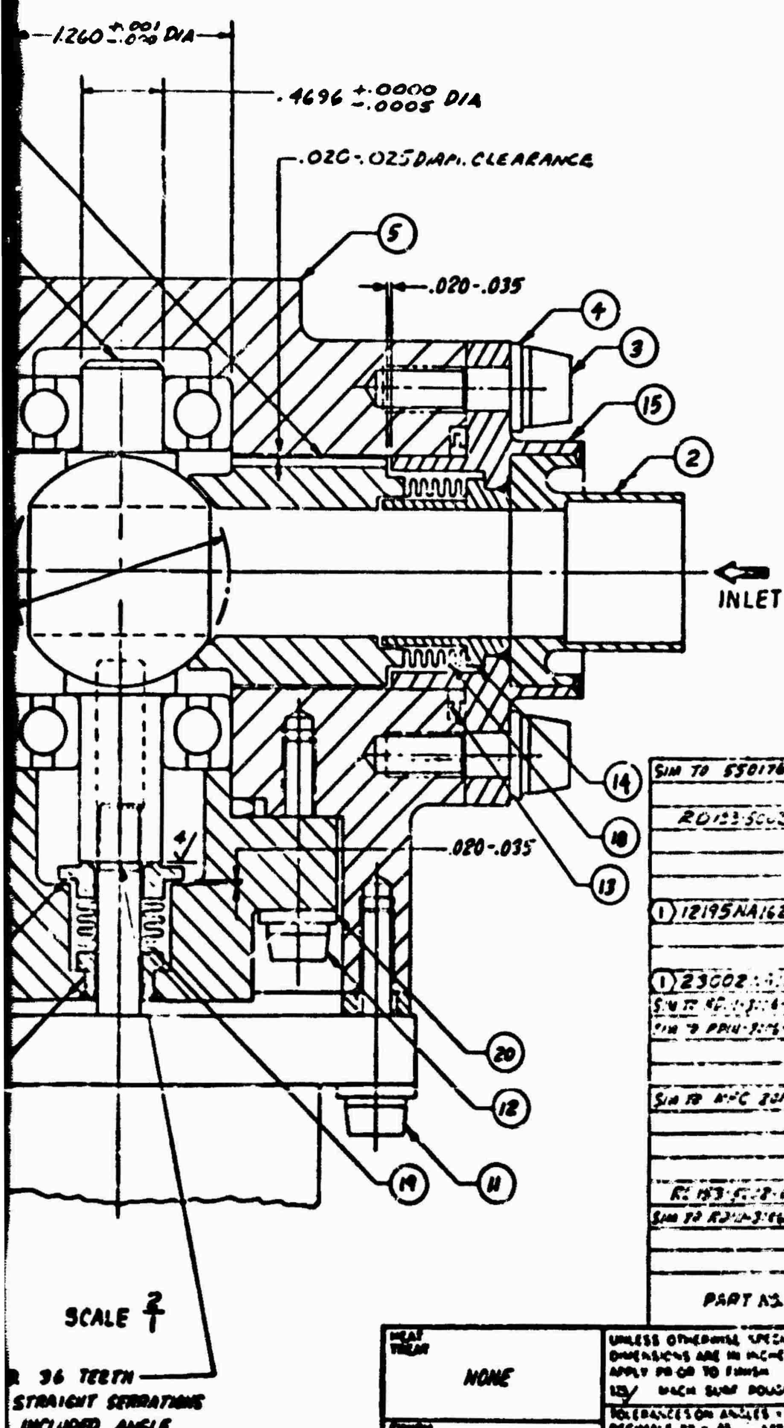
5. FLUID PRESSURE-
   OPERATING: 0 TO 2200 PSIA
   PROOF: 3350 ± 50 PSIA
4. FLUID HANDLED- HYDROGEN AT -200 TO +300F.
   WELD PER RA0107-027.
(2) ELECTRICAL RECEPTACLE PER MIL-C-27599.
1. TECHNICAL INDUSTRIES INC., HARRISON MFG. CO. DIV, BURBANK, CALIF.

NOTE: UNLESS OTHERWISE SPECIFIED

2

ACTUATOR CHARACTERISTICS	
RUNNING TORQUE	15 IN. LBS.
STALL TORQUE	60 IN. LBS.
MAX. TRAVEL	90°
SLEW RATE	1.0 SEC.
HORSEPOWER	.0036
DUAL POTENTIOMETER	
EACH POT.	0.3K TO 5K
FIXED PHASE VOLTAGE 110VAC 400~	

BELLOWS DATA	18	19
O.D.	1.000	.440
I.D.	.800	.310
NO. PLIES	3	3
MAT'L THICKNESS	.0025	.003
NO. CONVOLUTIONS	8	10
INSTALLED LENGTH	.375	.533
INSTALLED LOAD (Lb)	10.0	30.0
RATE (Lb/IN)	232.0	500.0



BEARING LOADS (LBS)	
THRUST	350
RADIAL	1500

PART NO.	ITEM NO.	NO. REQD	DESCRIPTION	MATERIAL
SIM TO 550176	22	1	NAMEPLATE	302 CRES
	21	1	CAP	INCO 718
RD153-5002-[illegible]	20	16	WASHER	A 286
	19	1	FORMED BELLOWS	INCO 625
	18	1	FORMED BELLOWS	INCO 625
	17	1	AC ELECT. ACTUATOR	COMMERCIAL
(1) 12195NA1625	16	1	K SEAL 12195	INCONEL X 750
	15	1	CAP	INCO 718
	14	1	RETAINER	INCO 718
(1) 23002-[illegible]250	13	1	K SEAL 12[illegible]50	INCONEL X 750
SIM TO [illegible]-1014	12	8	BOLT (10-32)	RENE 41
SIM TO PPIU-[illegible]-1012	11	2	BOLT (10-32)	RENE 41
	10	1	RETAINER	INCO 718
	9	1	SEAL	INCO 718
SIM TO [illegible]	8	2	BALL BEARING	STELLITE
	7	1	SHAFT	RENE 41
	6	1	SEAL	INCO 718
	5	1	HOUSING	INCO 718
RD153-5[illegible]-0004	4	6	WASHER	A 286
SIM TO R[illegible]-1818	3	6	BOLT (1/4-28)	RENE 41
	2	2	ADAPTER	321 CRES
	1	1	INSULATOR	MICA PLATE

HEAT TREAT	NONE
FINISH	NOTED
MAT.	NOTED

UNLESS OTHERWISE SPECIFIED DIMENSIONS ARE IN INCHES AND APPLY PRIOR TO FINISH
MACH SURF ROUGHNESS
TOLERANCES ON ANGLES
DO NOT SCALE PRINT

DR. J. L. MURDICK

Rocketdyne
North American Rockwell
Canoga Park, California

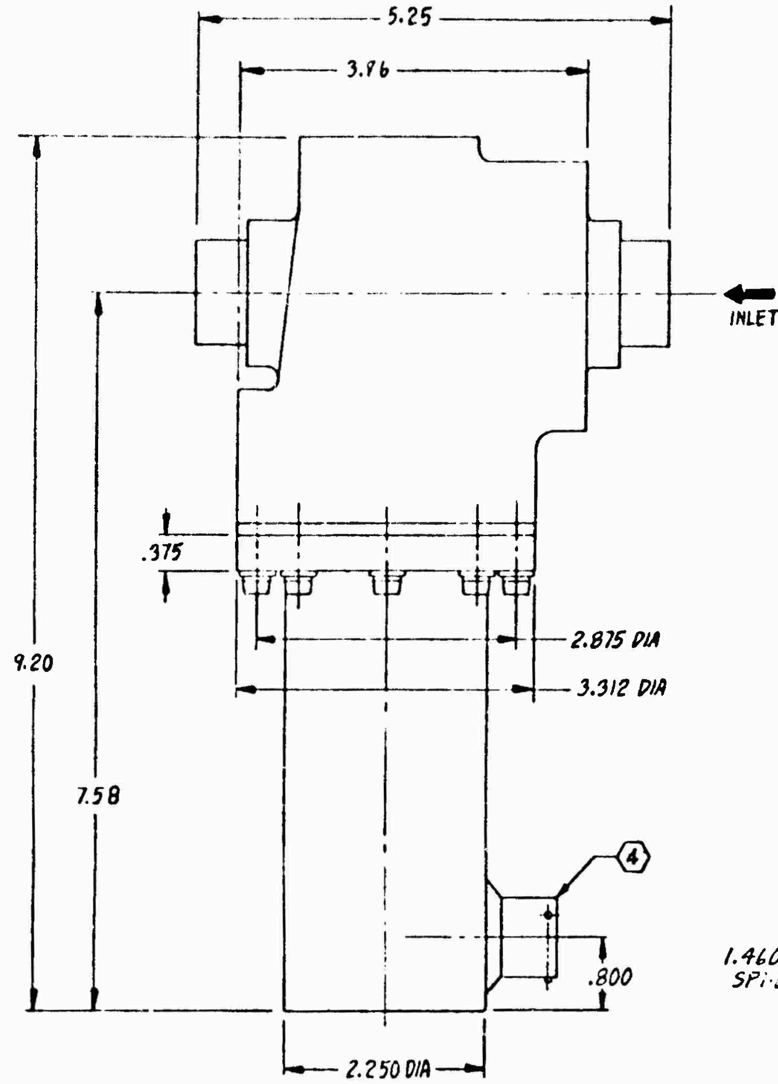
Figure 110. Turbine Inlet Control Oxidizer Valve

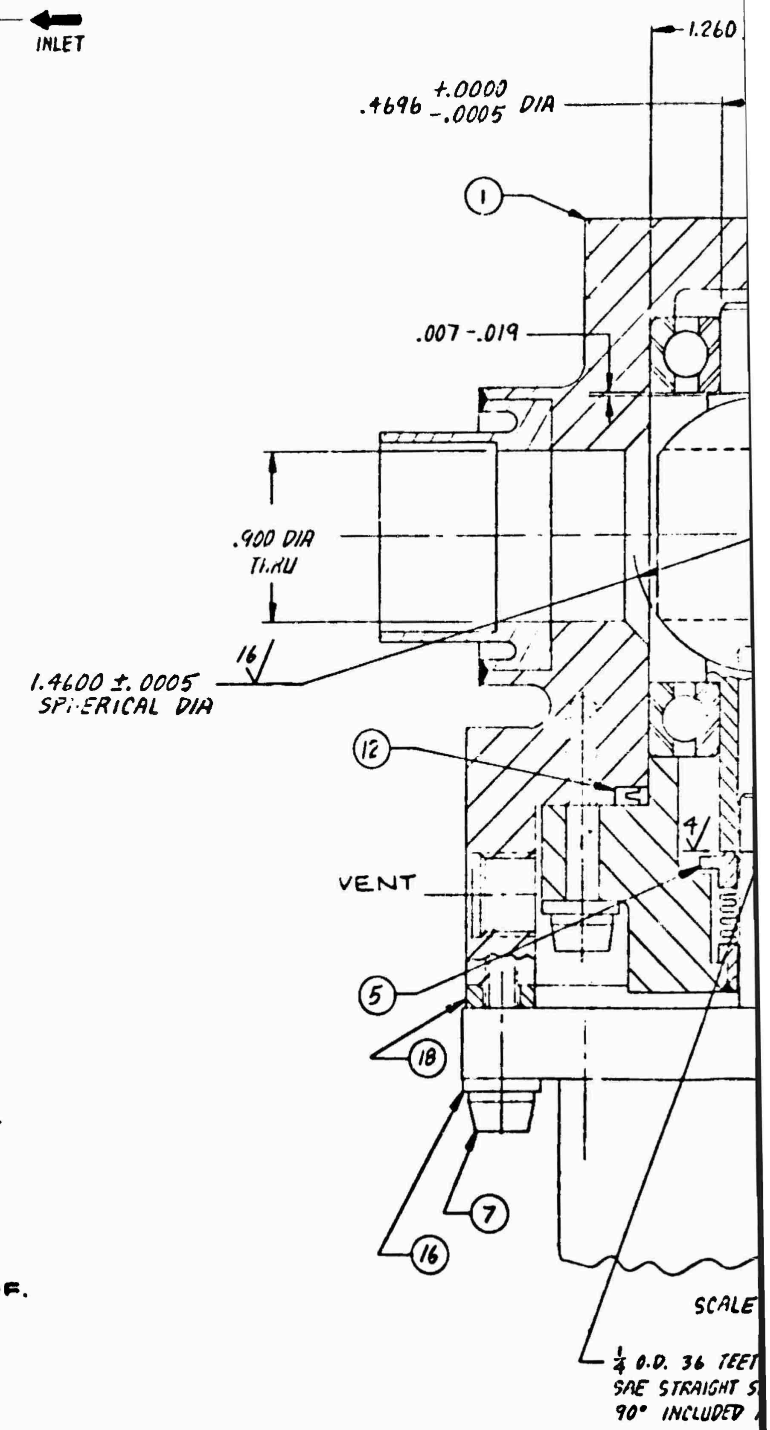
SIZE	CODE IDENT NO.	DRAWING NO.
D	02602	RS004978L

SCALE 1/1

BEARING LOADS	(LBS)
THRUST	350
RADIAL	2500

ACTUATOR CHAR
RUNNING TORQUE
STALL TORQUE
MAXIMUM TRAVEL
SLEW RATE
HORSEPOWER
DUAL POTENTIOMETER
EACH POT. 0.3
FIXED PHASE VOLTAGE



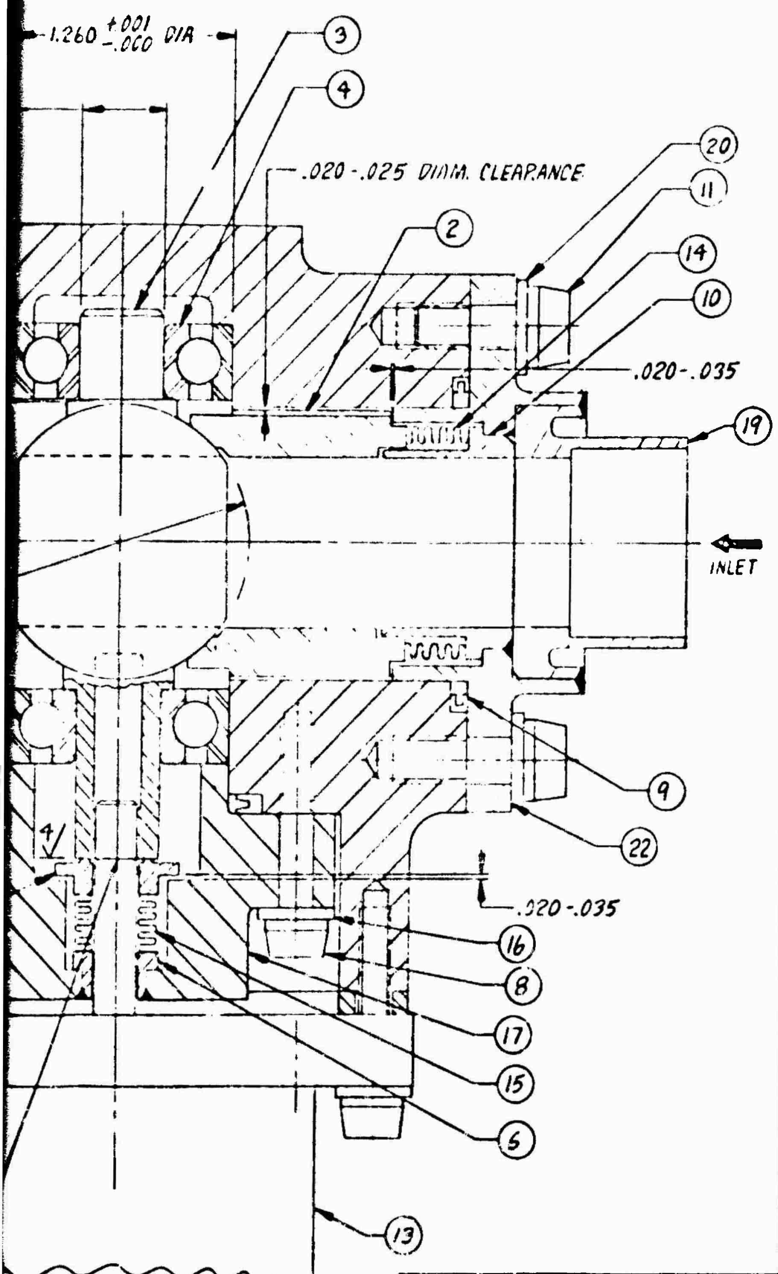


5 TECHNICAL INDUSTRIES INC. HARRISON MFG. CO. DIV. BURBANK CALIF
4 ELECTRICAL RECEPTACLE PER MIL-C-27599
3. WELD PER RA0107-027
2 FLUID PRESSURE -
OPERATING : 0 TO 2200 PSIA
PROOF : 3350±50 PSIA
1. FLUID HANDLED - HYDROGEN AT -200 TO +900F.

NOTE: UNLESS OTHERWISE SPECIFIED

CHARACTERISTICS	
...IE	15 IN. LBS
...E	60 IN. LBS
...EL	90°
	1.00 SEC
	.0036
...IOMETER	0.3K TO 5K
VOLTAGE	110 VAC 400~

BELLOWS DATA	14	15
O.D.	1.190	.440
I.D.	1.000	.310
NO. PLIES	3	3
MAT'L THICKNESS	.003	.003
NO. CONVOLUTIONS	10	10
INSTALLED LENGTH	.460	.533
INSTALLED LOAD (Lb)	15.0	30.0
RATE (Lb/IN)	375.0	500.0



SCALE 2/1

.36 TEETH
STRAIGHT SERRATIONS
INCLUDED ANGLE

ITEM NO.	NO. REQD	DESCRIPTION	MATERIAL
22	1	CAP	INCONEL 718
21	1	NAMEPLATE	302 CRES
20	6	WASHER	A286 CRES
19	2	ADAPTOR	321 CRES
18	1	INSULATOR	MICA PLATE
17	1	CAP	INCONEL 718
16	16	WASHER	A286 CRES
15	1	FORMED BELLOWS	INCONEL 625
14	1	FORMED BELLOWS	INCONEL 625
13	1	AC ELECT ACTUATOR	COML PART
12	1	K SEAL	INCONEL X 750
11	6	BOLT (1/4-28)	RENE 41
10	1	RETAINER	INCONEL 718
9	1	K SEAL	INCONEL X 750
8	8	BOLT (10-32)	RENE 41
7	8	BOLT (10-32)	RENE 41
6	1	RETAINER	INCONEL 718
5	1	SEAL	INCONEL 718
4	2	BALL BEARING	STELLITE
3	1	SHAFT	RENE 41
2	1	SEAL	INCONEL 718
1	1	HOUSING	INCONEL 718

HEAT TREAT	NONE
FINISH	NOTED
MAT'L	NOTED

UNLESS OTHERWISE SPECIFIED: DIMENSIONS ARE IN INCHES AND APPLY PRIOR TO FINISH
125/ MACH. SURF. ROUGHNESS
TOLERANCES ON ANGLES ± 0° 30'
DECIMALS .XX ± .03 .XXX ± .010
HOLES NOTED "DRILL"

OVER	THRU	TOLERANCE	
.0000	.0400	+ .0015	— .0010
.0400	.1300	+ .0030	— .0010
.1300	.2290	+ .0045	— .0010
.2290	.5000	+ .0060	— .0010
.5000	.7500	+ .0070	— .0010
.7500	1.0000	+ .0090	— .0010
1.0000	2.0000	+ .0120	— .0010

DO NOT SCALE PRINT

DRN | DATE
CHK | DATE
DSGN
MATL
STRUCT
DESIGN ACTIVITY APVD | DATE

Rocketdyne North American Rockwell — Canoga Park, California; McGregor, Texas

Figure 111. Turbine Bypass Control Valve

SIZE D | CODE IDENT NO. 02602 | DRAWING NO. RS004979L

SCALE 1/1 | SHEET

The shaft seal is a rotary face seal welded to a sationary, short-stroke, hydroformed bellows. The shaft seal is on the low-pressure side of the throttle valve so that leakage is reduced to a very low level. The leakage past the shaft seal is routed overboard.

The throttle valve is positioned by means of an electrical rotary actuator. The actuator is very compact, utilizing an electric motor and a planetary gear system with open and closed stops and dual position potentiometers for feedback and telemetering. The electric motor is an a-c servomotor that requires no brushes to avoid radio frequency interference and no permanent magnets that may weaken at high temperatures. The actuator is coupled to the throttle shaft through a spline. There is an additional seal on the actuator shaft to prevent hot gas from seeping into the motor.

The ball seal serves a dual purpose. In the closed throttle position, the ball seal reduces the flow to an acceptable level. In the semi-open position, the seal forms part of the throttling area. The ball seal is also welded to a stationary, short-stroke, hydroformed bellows which, in turn, is protected by a shield from the eroding effects of a high-velocity, hot-gas stream. Both the shaft seal bellows and the ball seal bellows are protected from overdeflection during the assembly procedure, or possibly during valve operation transients.

Heat transfer from the valve into the electrical actuator was reduced to a tolerable level by minimizing the heat transfer area and providing an insulator between the valve housing and the electrical actuator base.

The design utilizes high-nickel, heat-resistant alloys with matching coefficients of expansion. Sufficient clearances were provided between moving parts, including the ball bearings, so that during the initial moments of hot firing, thermal gradients will not cause the valve to bind. Sealing surfaces are flame-plated to ensure minimum wear, low friction, antigalling characteristics, and good surface finish. Valve torque was reduced by throttling at the ball inlet only, and allowing the ball outlet to be open to downstream pressure.

Maximum valve torque, including friction, is approximately one-fourth of the stall torque of the electrical actuator. This ratio provides sufficient acceleration capability for adequate transient response when the throttle valve is part of a closed servoloop.

The valve designs permit ease of assembly and disassembly with no special tools required.

Main Fuel and Oxidizer Valves.

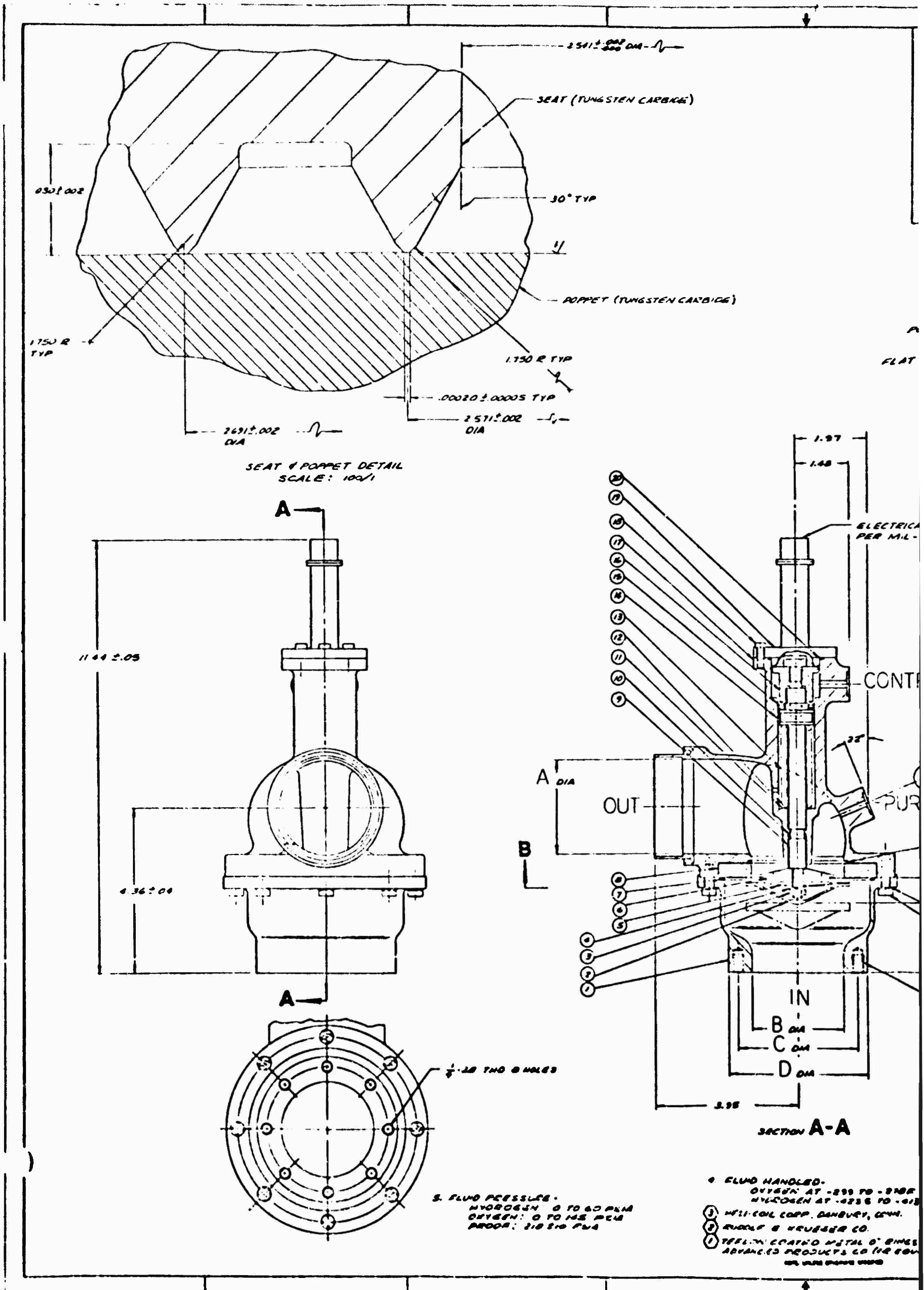
The main propellant valves provide on/off flow control of oxidizer and fuel and are located upstream of the turbopumps. A single valve design was selected to accommodate both fuel and oxidizer applications, and is shown in Fig.112. A tabulation of inlet and outlet diameters, which are different for the fuel and oxidizer valves, also is shown on the layout.

A poppet valve design was chosen to simplify construction, thus ensuring the least cost and highest reliability commensurate with the least number of detail parts. The valve is held in the closed position by a helical spring and by pressure acting on the seat unbalanced area.

With the poppet located upstream as shown on the layout, the only closed-position internal leak path is the seat-to-poppet interface, thus requiring no low leakage shaft seals. The valve is opened by applying 750 ±50 psia helium to the control port, at which time the differential pressure across the piston overcomes the helical spring and propellant pressure forces. As the control pressure is vented, the helical spring and differential propellant pressure across the poppet cause the valve to close.

Detail design features include:

1. Removable, self-aligning tungsten carbide seat and poppet configuration as developed under NASA Contract NAS3-14350 for extended cycle life (10^6) cycles and low leakage (o.1-scim helium) at temperatures down to -250 F.


SEAT (TUNGSTEN CARBIDE)
30° TYP
POPPET (TUNGSTEN CARBIDE)
1.750 R TYP
SEAT & POPPET DETAIL
SCALE: 100/1
A
B
OUT
IN
SECTION A-A

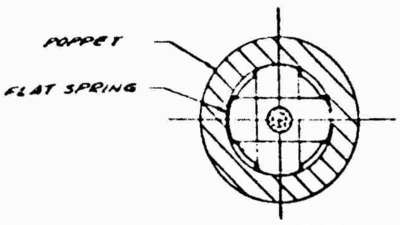
2

HELICAL SPRING DATA

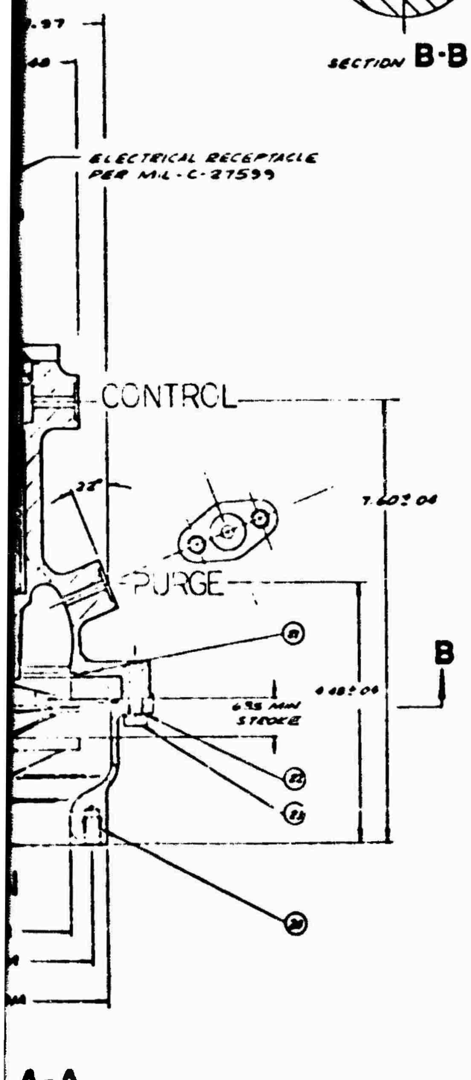
O.D.: .830
WIRE DIA: .140
NO. ACTIVE COILS: 7.5
NO. TOTAL COILS: 9.5
RATE: 236 LB/IN
INSTALLED LOAD: 50 LB AT 2.130 HT

FLAT SPRING DATA

THICKNESS: .022
WIDTH: .750
O.D.: 1.860
INSTALLED LOAD: 15 LB AT .053 HT



SECTION B-B



SECTION A-A

DASH NO	PROPELLANT	A DIA	B DIA	C DIA	D DIA
001	FUEL (H_2)	2.500	2.500	3.250	3.800
011	OXIDIZER (O_2)	2.700	3.000	3.750	4.300

	NO	QTY	DESCRIPTION	MATERIAL	FINISH
3	24	8	INSERT, HELICOIL	303 CRES	SILVER PLATE
	23	8	SCREEN	A286 CRES	-
	22	8	WASHER	A286 CRES	-
1	21	1	SEAL	321 CRES	TEFLON COAT
1	20	1	SEAL	321 CRES	TEFLON COAT
	19	1	TRANSDUCER	CRES	-
	18	3	SCREEN	A286 CRES	-
	17	1	BODY	347 CRES	-
	16	1	STOP	A286 CRES	-
2	15	1	DELTA SEAL	TEFLON, CRES	-
	14	1	BUSHING	TEFLON, TFE	-
	13	1	SPRING	ELGILOY	-
	12	1	WASHER	303 CRES	-
	11	1	GUIDE	TEFLON, TFE	-
	10	1	BUSHING	TEFLON, TFE	-
	9	1	PISTON	A286 CRES	-
	8	1	SEAT	TUNGSTEN CARBIDE	-
1	7	1	SEAL	321 CRES	TEFLON COAT
	6	1	LIFTER	A286 CRES	-
	5	1	SPRING, FLAT	BERYLLIUM COPPER	-
	4	2	WASHER	303 CRES	-
	3	1	NUT, LOCKING	A286 CRES	SILVER PLATE
	2	1	POPPET ASSY	TUNGSTEN CARBIDE & NI. BST. (BRAZED)	-
	1	1	FLANGE	6061-T651 AL ALLOY	ANODIZE

-AT -.299 TO -.310F
-N AT -.423F TO -.413F
-., DANBURY, CONN.
-WEBER CO
TO METAL O-RINGS FROM
-DUCTS CO (OR EQUIVALENT)

NOTED
NOTED
NOTED

Figure 112. Main Fuel and Oxidizer Valve

E 02602

2. Teflon bushings on the piston head and shaft and on the spring ID eliminate metal-to-metal sliding wear.
3. Spring-loaded Teflon dynamic piston seal to prevent gross helium usage during valve opening cycle
4. Position transducer for monitoring valve response and position
5. Welded outlet flange and bolted inlet flange

COMBUSTION WAVE IGNITION SYSTEM

An evaluation of candidate ignition systems was conducted, as reported in Appendix D. The combustion wave ignition system was selected as the most favorable ignition method for either the double- or single-panel aerospike engine application. Primary reasons for this selection were: (1) current technology status, and (2) minimum complexity in integrating this system into the engine.

The combustion wave igniter concept utilizes a spark-induced combustion wave passing through an unburned, gaseous oxygen/hydrogen mixture to ignite a pilot element at the main injector face. The combustion wave is initiated by an electrical arc discharge spark in a premix chamber. The resultant combustion wave begins to propagate in the unburned mixture in the direction of flow. Compression, shock, and eventually a detonation wave develops in the unburned mixture. A flow schematic for this ignition system is shown in Fig.113.

As presently conceived, the combustion wave ignition element is a set of triaxial tubes that are flush-mounted in the main injector face of each of the segments. The core of the triaxial element is the combustion wave tube, and the annuli form the pilot element that is ignited by the combustion wave. The combustion wave for any number of these elements is supplied from a central premix chamber equipped with dual integrated spark plug/exciter units. A preliminay design of a premixer with propellant control valves is shown in Fig.114. Its primary function is to prime the combustion-wave tubes with a combustible mixture.

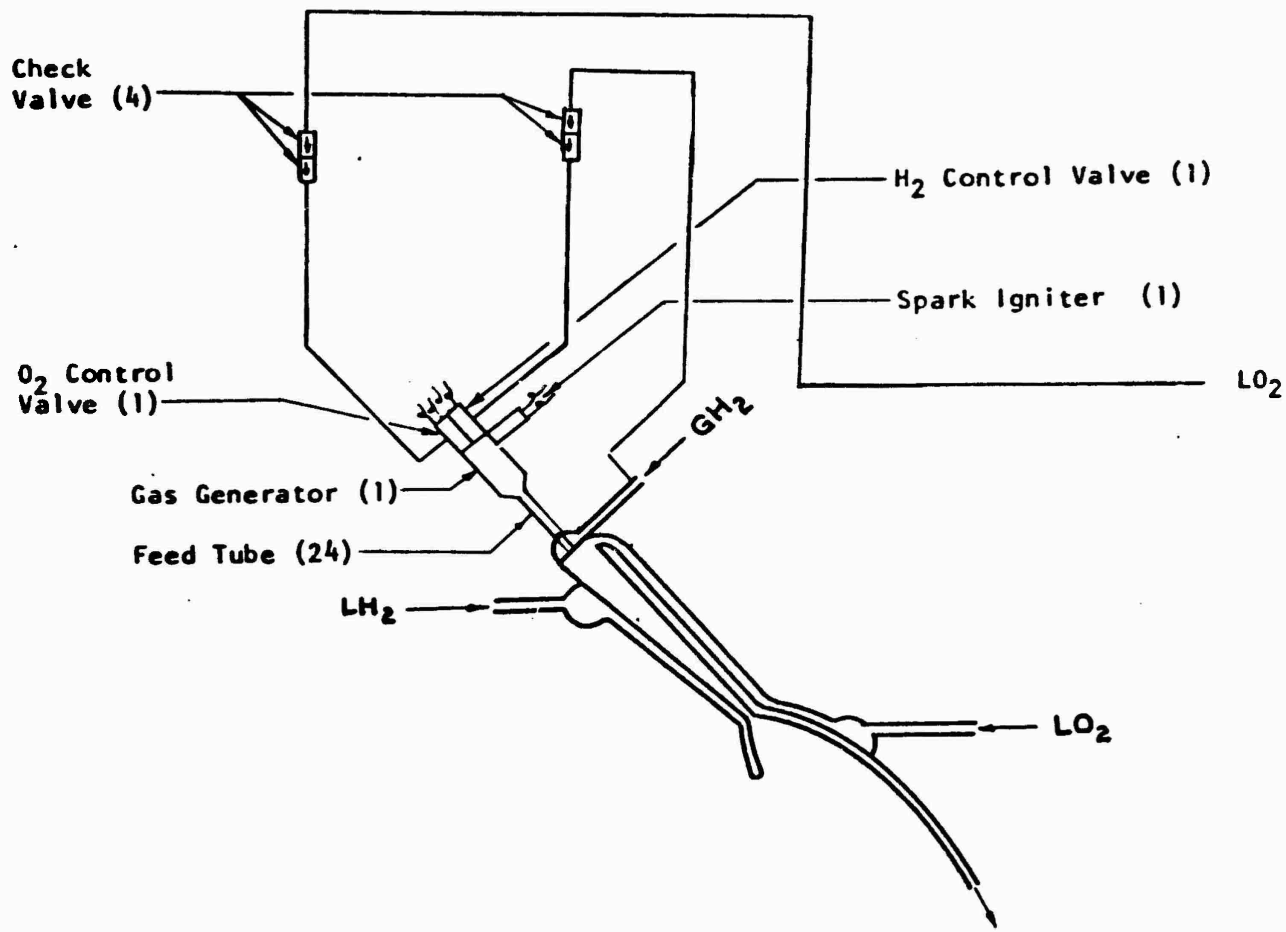


Figure 113. Combustion Wave Igniter Flow Schematic

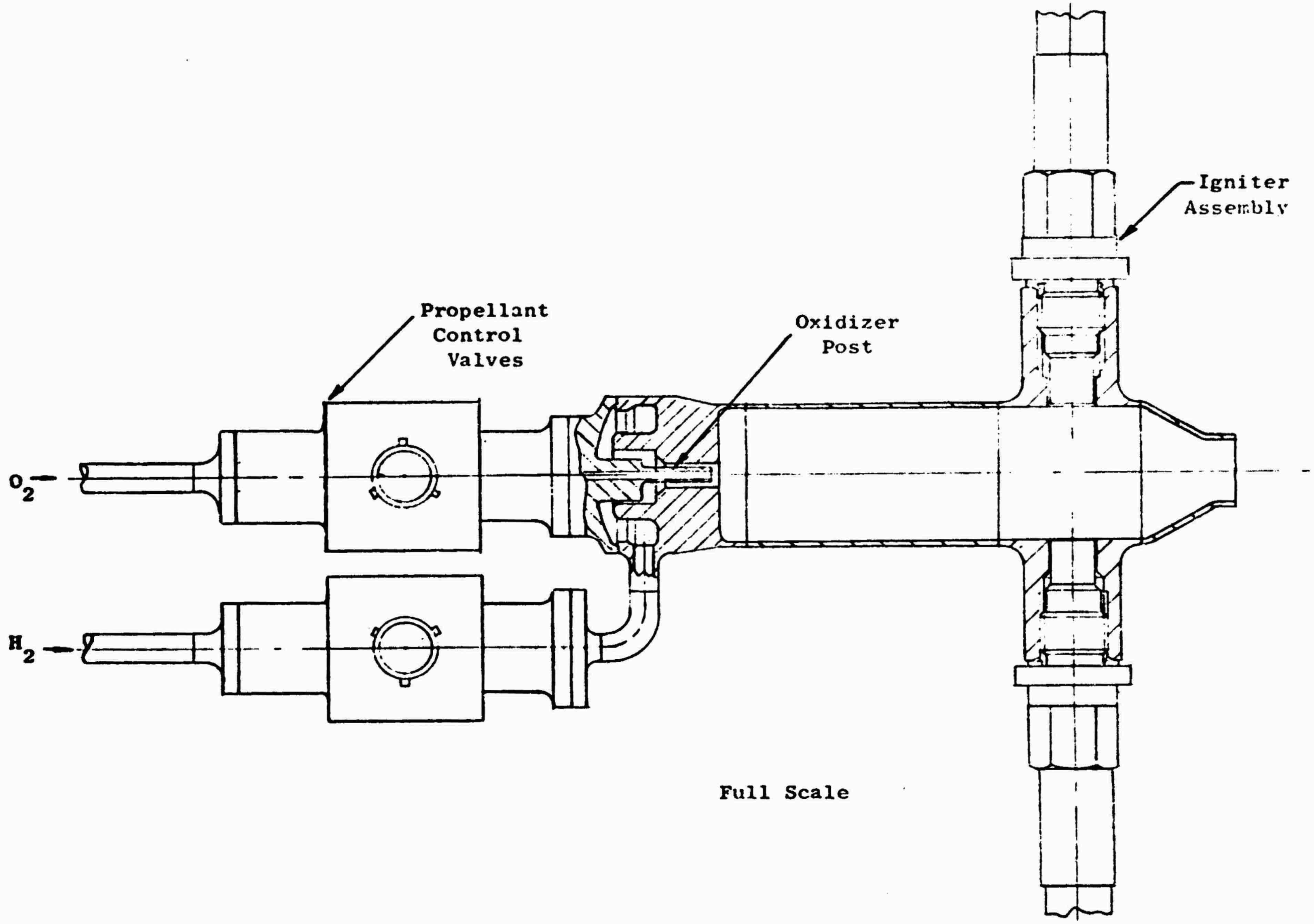


Figure 114. Combustion Wave Ignition Premix Chamber

The ignition energy is supplied to the premix chamber by redundant integrated plug/exciter units. The premix chamber feeds the mixed, unburned propellants to the annular combustion wave delivery manifold. The 24 triaxial elements, one for each segment, are then fed from the manifold.

Fuel and oxidizer are supplied to the premix chamber under tank-head pressure during the initial portion of the start transient. The fuel is extracted from the fuel supply line between the turbine exhaust and the thrust chamber fuel inlet manifold. The oxidizer is extracted at the oxidizer pump discharge and passes through the oxidizer heat exchanger that is also necessary to gasify oxygen for main tank pressurization. Check valves are provided in the premix chamber propellant supply lines to prevent backflow into the propellant supply lines at the time of ignition of the combustion wave. Pilot flows in the annuli around the combustion wave tube are supplied from the main injector manifolds.

This tank-head propellant feed method for the combustion wave ignition was selected based on the design selected in the linear thrust chamber breadboard engine program.

Several experimental programs are in progress, as discussed in Appendix D, investigating combustion wave ignition systems with low-pressure propellant supply. However, further work should be conducted to ensure that reliable ignition will occur with a wide range of initial propellant and hardware temperatures along with the low pressures. The requirement that the engine be capable of an immediate or short-term restart, or a start after an extended orbital coast will result in a range of propellant inlet conditions and initial hardware temperatures. The mixture ratio of the premixed propellants in the combustion wave delivery tubes is critical to reliable operation of this system. Because of this, the uncertainty associated with mixed-phase flow conditions which may exist under tank-head operation suggests that gaseous propellant storage bottles may be required to provide reliable ignition over the full range of tank-head start conditions. The storage pressure is not critical, however, and reliable operation of combustion wave ignition at 1 atmosphere has been demonstrated. Also, in relation to the uncertainty in propellant inlet conditions, a single combustion wave is considered sufficient to

ignite all of the segments but, due to this possible range of inlet propellant conditions, both to the premix chamber and to the pilot elements in the thrust chamber segment, a series of waves may be necessary to achieve the required reliability. The exact number of waves generated would be a subject for investigation. If tank-head start operation cannot be adequately demonstrated, refillable gaseous-propellant storage bottles may be added to the system. The bottles would be refilled during engine operation and supply propellants during start in the blowdown mode. The oxidizer bottle would be filled with gaseous oxygen from the heat exchanger at the oxidizer turbine discharge. The fuel bottle is supplied with heated hydrogen after it passes through the turbines. A preliminary analysis was conducted to determine system weight effects (see Appendix D) if storage bottles are added. A possible weight increase of 41.5 pounds would result from the addition of the storage bottles.

The sequence for the combustion wave ignition system is as follows:

1. At engine start signal, the engine main fuel valve is opened, followed by opening of the engine main oxidizer valve. The igniter element pilot manifolds are primed with propellants flowing to the thrust chamber.
2. The premix chamber valves are then opened and the combustion wave tubes are primed.
3. Upon expiration of an ignition delay timer, the spark plug is fired and the premix chamber oxidizer valve is closed. The combustion wave propagates to the injector face and ignites the pilot flows in each segment.

The individual combustion wave elements are positioned at the centers of the segment injectors and are only slightly larger than the primary elements. During ignition, the propellant flow to the pilot element surrounding the combustion wave port) will have a mixture ratio of approximately 2:1 or greater to ensure a hot enough flame to ignite the remaining primary injector elements. During mainstage, the igniter pilot element mixture ratio will be the same as the primary injector elements to minimize potential performance losses and prevent temperature profile maldistributions.

The premix chamber is designed to prime the mixer and combustion wave delivery tubes with unburned gaseous propellants at a mixture ratio of approximately 2.5:1. The total premixer flow prior to ignition is approximately 0.007 lb/sec, or 0.00029 lb/sec per segment, which is approximately 1.5 percent of the segment primary injector flow. After ignition, the premixer oxidizer valve is closed and gaseous hydrogen is allowed to flow through the delivery tubes to each segment for the duration of mainstage operation. The gaseous hydrogen supply pressure continues to rise with pump discharge pressure so that the pressure within the premixer is always above the segment combustion chamber pressure, and flow is always in the direction of the combustion chamber segment. The total combustion wave ignition system weight is estimated at 7 pounds.

ENGINE DEVELOPMENT PROGRAM

PROGRAM PLANS

The development program plans for the aerospike engine systems are presented herein. Because the single- and double-panel designs both operate at the same thrust level and utilize the expander topping turbine drive cycle, the development program plans for both systems will be identical. The increased complexity resulting from the slightly more intricate design features of the double-panel cooling circuit will not result in a change in the program plan; however, fabrication costs for the thrust chamber will be affected. The plans delineate the development requirements for a complete engineering development program, a demonstrator engine development program, and production of the baseline design, considering the three applications of: (1) ground based-reusable type, (2) ground based-expendible (HEUS) type, and (3) space based-reusable type.

COMPLETE ENGINEERING DEVELOPMENT PROGRAM

The complete engineering program entails development through flight certification, achieving engine readiness for production and field operations. The program is directed toward minimizing the risks involved in any major development effort by verifying that the engine design meets the limits of the design requirements at the lowest hardware assembly level, conducting extensive overstress testing to identify marginal conditions, and continuous monitoring of critical parameters to provide visibility for assessing progress and timely identification of problem areas. The overstress testing will consist of functional testing of components, subsystems, and engine assemblies beyond the 3-σ operational limits (operating range encompassing 99 percent of hardware variation), as well as structural loading to failure of selected components. The functional testing beyond the 3-σ operating limits will be conducted so that the load-strength relationship at critical locations in the test hardware corresponds to conditions of maximum operational loading of an acceptable minimum strength part.

The program includes verification of design requirements, operational modes, and ground support equipment resulting in a well-integrated development effort. Special emphasis would be placed on verifying characteristics such as stability and suitable system dynamics (Pogo). An integral part of the development effort is the early identification of potential failure modes and preparation of detailed contingency plans to ensure early solution of problems and thus minimize impact on the program. The early resolution of potential problems is enhanced by maintaining a "frozen baseline design" concept where changes are implemented only for the resolution of problems, excluding "nice to have" types of marginal improvements, and by verifying that the design meets the specified requirements at the lowest practical hardware assembly level as early in the program as possible. The verification of each requirement is conducted based on a selected "verification complete" criteria. Verification of a requirement is complete as soon as the stated objectives have been achieved and the development effort is continuously shifted to those requirements yet to be verified resulting in a cost-effective program.

Flight certification consists of approval by the procuring agency that the engine design has met the specified requirements and is ready for operational use. The flight certification is based on satisfactory verification of all requirements followed by a formal flight certification demonstration conducted under the cognizance of the procuring agency. Certification demonstration of each requirement is conducted after it has been verified that the engine design meets the limits specified and is initiated at a convenient point in the program. The service life capability of the design is certified based on the cumulative life history of all the development hardware and a final life demonstration to be conducted simulating a typical mission duty cycle.

Ground Based-Reusable Engine Design

The development program consists of 55 months of design, fabrication, and test effort. The design and fabrication schedules are shown in Fig.115 and the test program is presented in Fig.116. The production release point and the subsequent first engine delivery also are shown in Fig. 116.

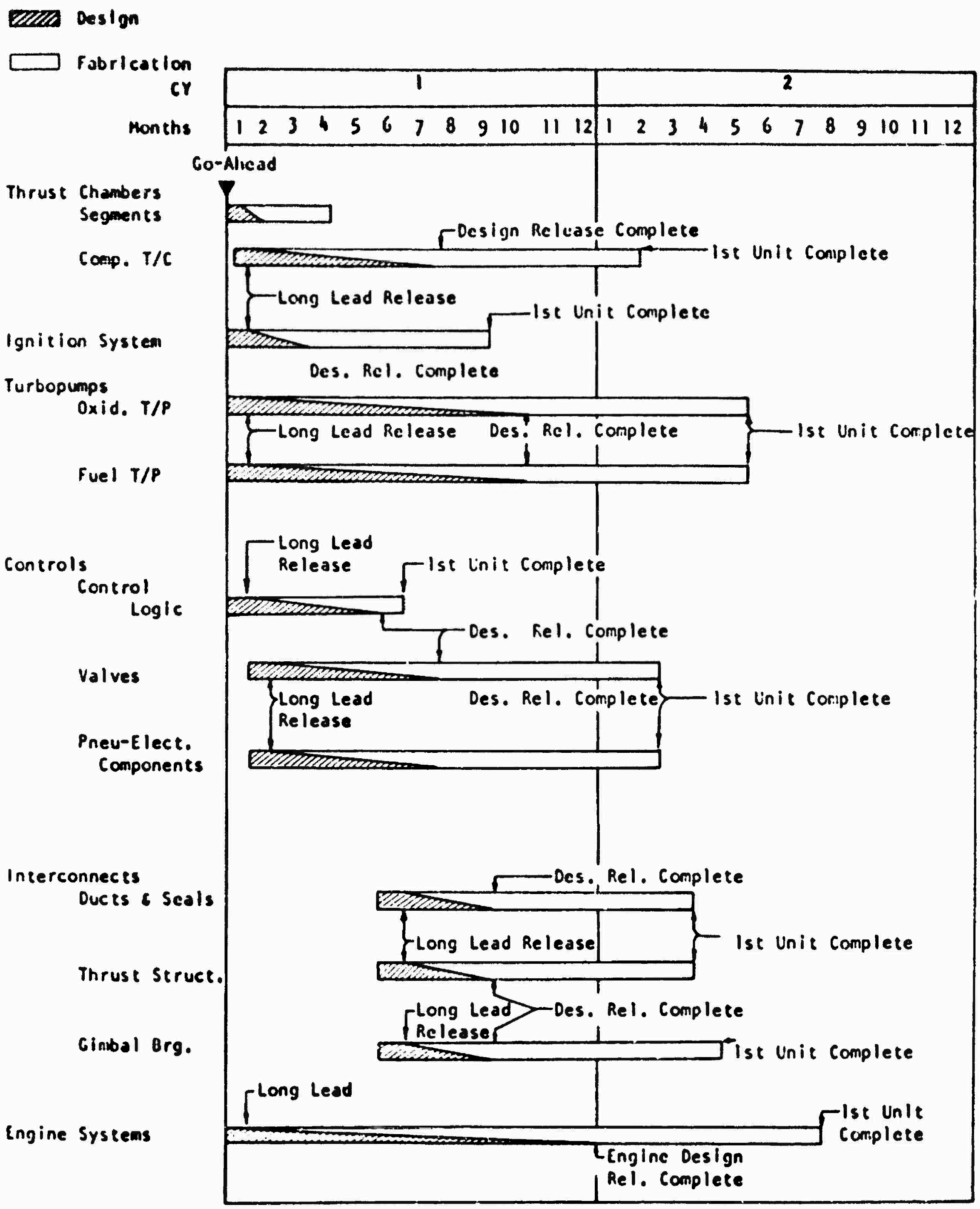


Figure 115. Complete Engineering Design and Fabrication Schedule (First Unit Delivery)

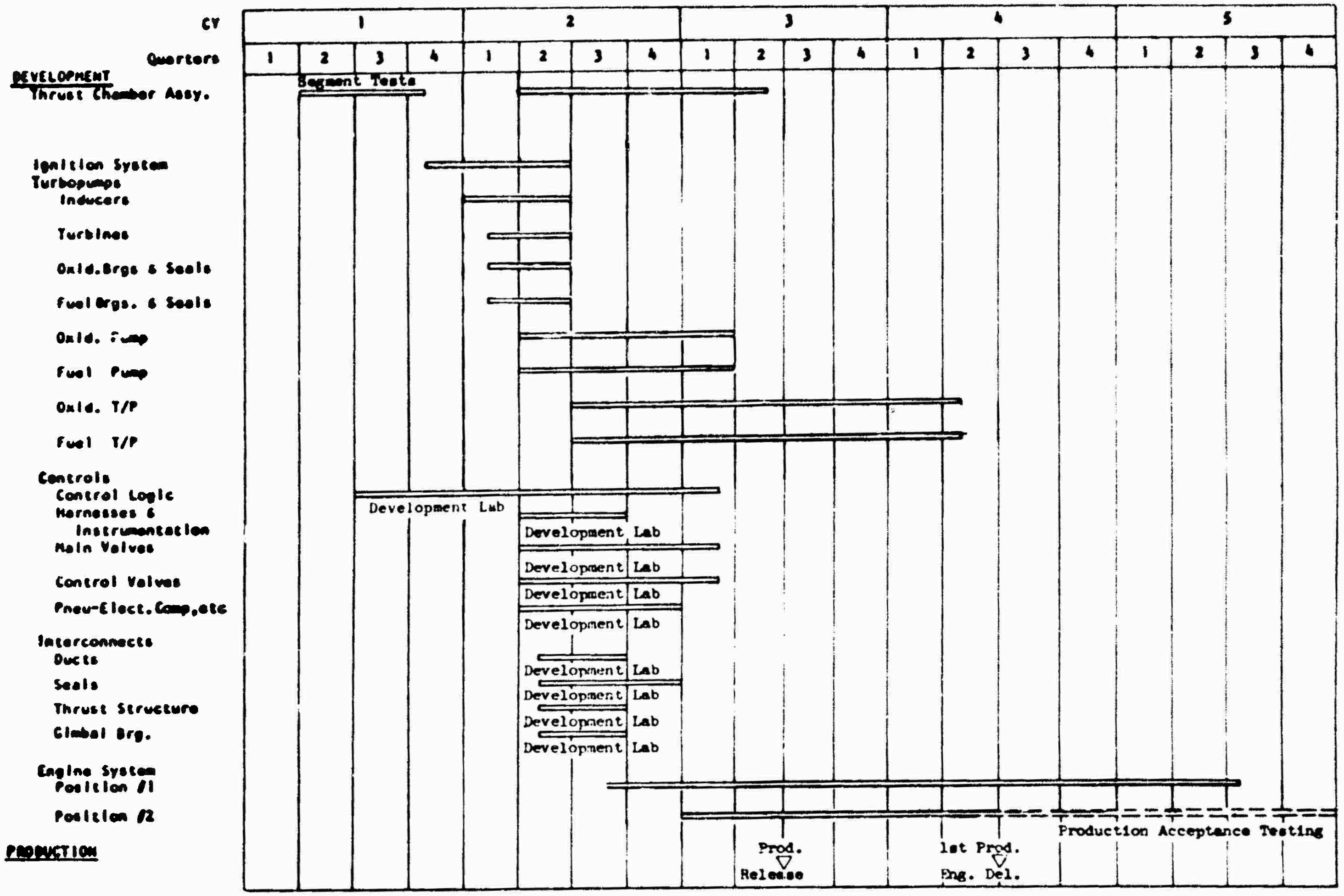


Figure 116. Program Test Plan - Complete Development (Aerospike Engine - Reusable)

The design and fabrication schedules are geared to early release of long-lead material procurement and sequential drawing releases to support the fabrication process to meet the early hardware delivery schedules. Emphasis is placed on fabrication and testing of the specific baseline hardware configuration, as opposed to workhorse configurations, with the exception that breadboard electrical hardware will be initially utilized for establishing the control logic unit component requirements.

The significant advantage of segmentation for aerospike thrust chamber construction over other configurations will allow early initiation of major subsystem testing. This testing will provide early demonstration of critical design features of the combustion chamber and injector at a significantly reduced cost as well as reduce the overall thrust chamber development effort. The extended thrust chamber and turbopump testing shown in Fig. 116 is primarily directed toward developing the significant service life capability specified as a requirement of the system.

The control system component testing time phasing, particularly with respect to the propellant and control valves, is planned to be consistent with major subsystem and engine testing in lieu of initiating the effort early. This approach minimizes the possibility of duplicate valve testing resulting from subsequent engine baseline design modifications.

Because engine system testing is a significant cost driver in the development of a rocket engine, effective utilization of component and subsystem testing for verifying design requirements will reduce the total engine system testing by eliminating those failure modes that can be resolved at a lower hardware assembly level, and thus result in a more cost-effective program.

To accomplish the desired program efficiency, emphasis is placed on the following:

1. Each requirement and its verification is identified.
2. Requirement verification is performed at the lowest hardware level practicable, and as early in the program as possible.
3. Overstress testing is used to accelerate failure mode detection at the component, subsystem, and engine levels.

Based on these considerations, a total of 23 complete engine buildups (6 new engines) was selected for the development program. The length of the overall development effort and the number of engine system tests were established based on historical design, fabrication, and test experience. A total of 970 engine tests were estimated to be necessary to achieve program objectives. The resulting time required for achieving the intended number of tests is 35 months (Fig.117). The 35-month engine test period, together with a 20-month design, fabrication, and preliminary component and subsystem test period, results in an estimated 55-month development program.

Because of the major emphasis on reusability, the engine test program is geared to exposing a selected sample of engines to significant number of recycles in keeping with the long life requirements of the program. A recycle consists of detailed teardown, inspection, and reassembly (with the original components if sufficient life remains) following a test series of up to 300 thermal cycles.

Ground Based-Expendible Engine Design

Utilizing the Ground-Based Reusable Development Program as a point of reference, the limited life requirements of the ground-based expendible engine design will result in a significant reduction in the portion of the development testing that is directed toward verifying the life requirements. The major change is a reduction in the life testing conducted at the subsystem level (thrust chamber assembly), turbopump assemblies, etc.) carried on concurrently with the engine system hot-fire test effort.

Though the limited life requirement will impact the specific design features of the engine in that a less conservative design policy can be adopted, the design and fabrication schedule will not significantly change over that shown for the reusable-type engine design. However, by eliminating that portion of the engine test effort primarily directed toward verifying the long- life requirement, a reduction in the test program (Fig.117) of approximately 20 percent is achieved. The development test program for the expendible design and the initial production schedule are shown in Fig.118 The expendible design of the engine requires only

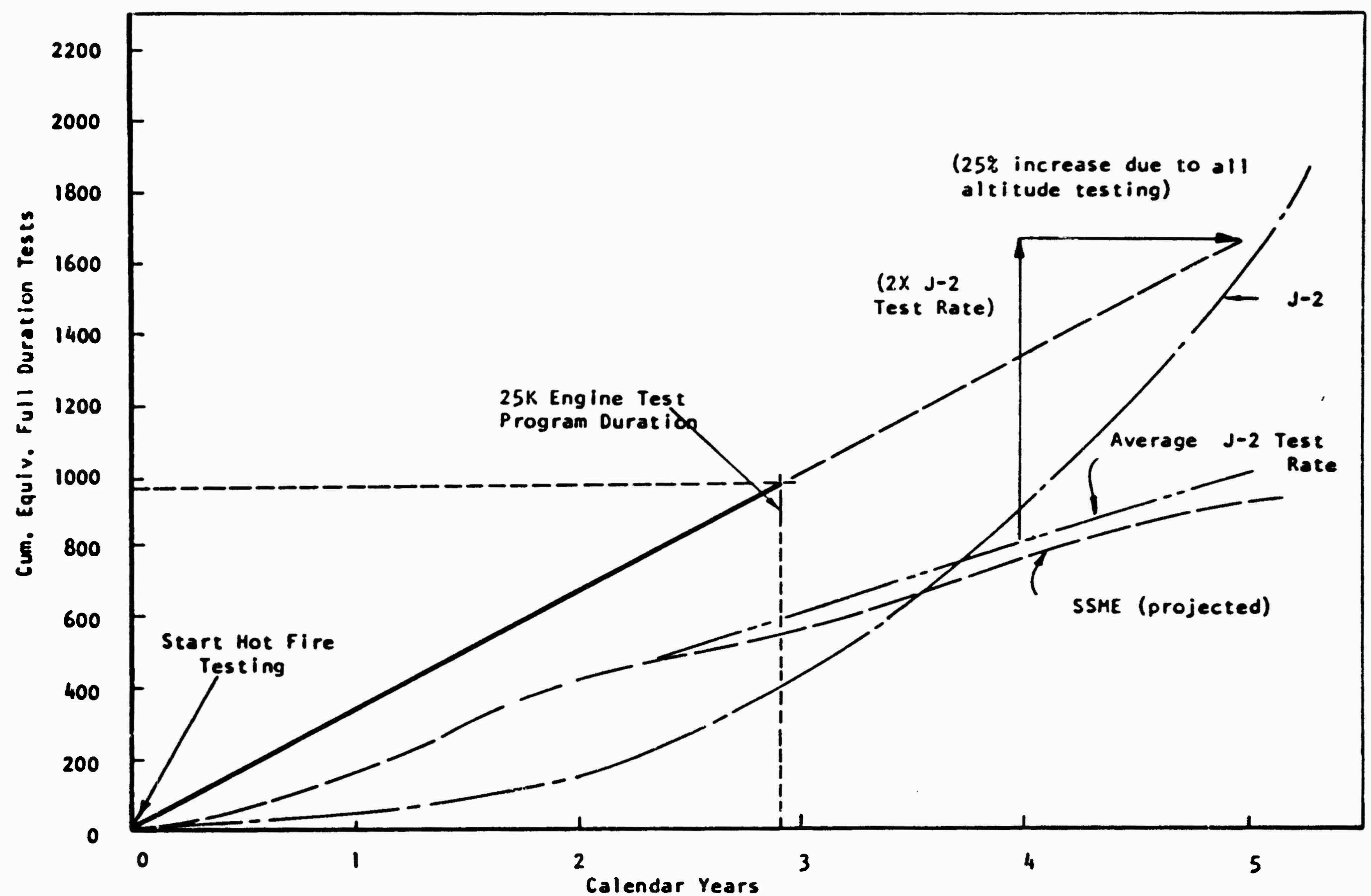


Figure 117. Projected Engine System Test Program Duration

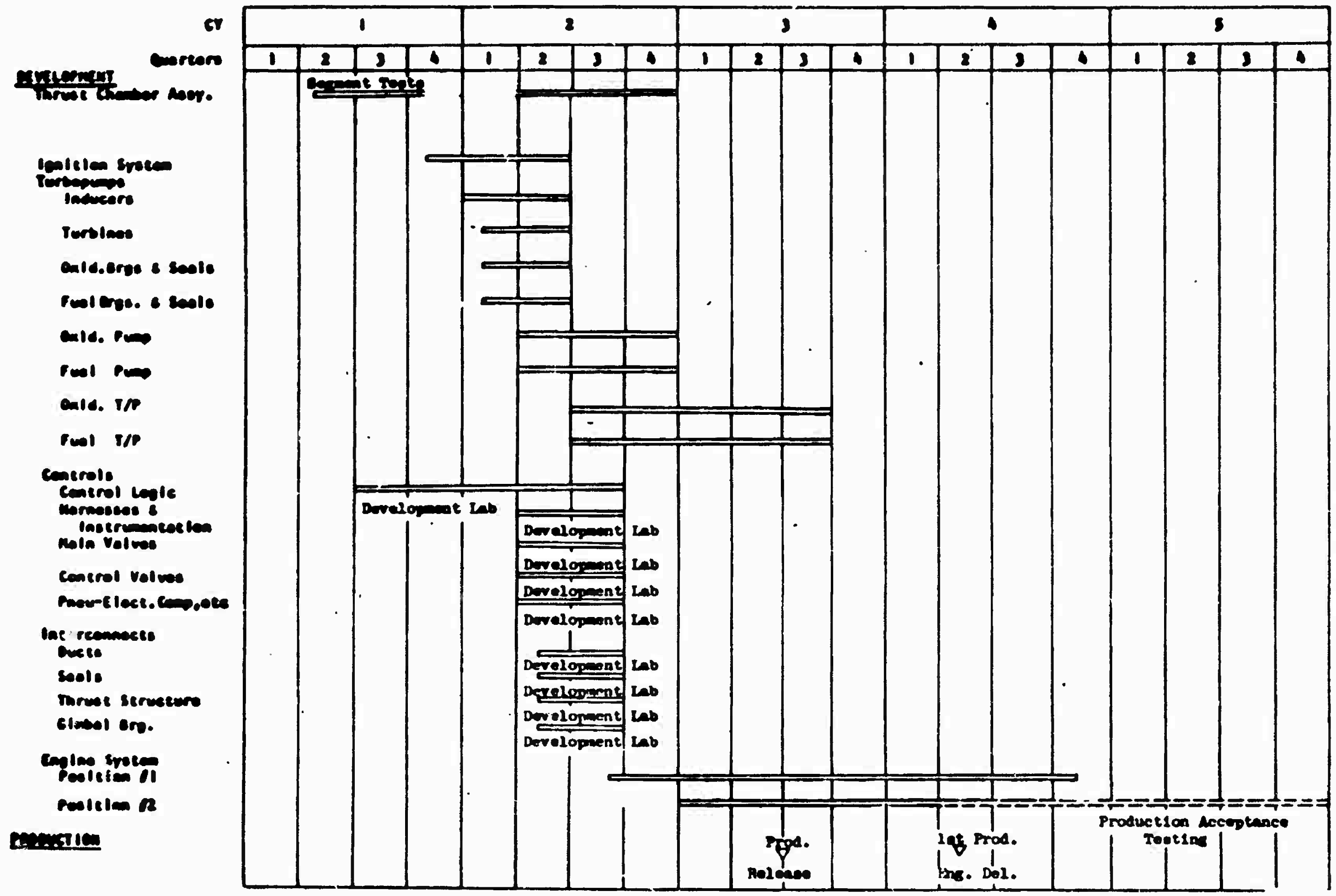


Figure 118. Program Test Plan - Complete Development
(Aerospike Engine - Expendible)

a limited life capability and, therefore, greater emphasis is placed on testing new hardware. The average recycle cost is increased due to more hardware replacement during each recycle and the number of recycles of hardware has been reduced.

Spaced-Based Reusable Engine Design

The space-based development effort will parallel the Ground-Based Reusable Engine Development Program with the addition of extensive laboratory simulation of space maintenance conditions. The proposed space maintenance approach is presented subsequently under the discussion of maintenance concepts. The overall design, fabrication, and test requirements remain as shown for the ground-based concept.

DEMONSTRATOR ENGINE DEVELOPMENT PROGRAM

The demonstrator engine program represents a cost-effective development approach for demonstrating feasibility of the concept and establishing a basis for assessing the service life capabilities of the engine design. To achieve these goals, the scope of component and subsystem level development has been restricted to demonstration of basic operability as compared to full-scale development in a complete engineering development program. The design and fabrication schedules to first unit delivery for test are essentially as presented for the complete engineering program (Fig. 113) because the flow time for the first unit would remain the same. However, the magnitude of effort required to support the design and fabrication of the limited hardware requirements would result in a significant reduction in magnitude of effort. The overall demonstrator engine development program consists of 31 months with the test portion as shown in Fig. 119.

As specified for the complete engineering development program, early segment testing would be utilized to identify critical problems to allow for implementation of corrective action in the fabrication of the initial thrust chamber assemblies. The turbopump assembly design features are tested over the full range of operation, but only to the extent necessary to establish suitability for engine testing.

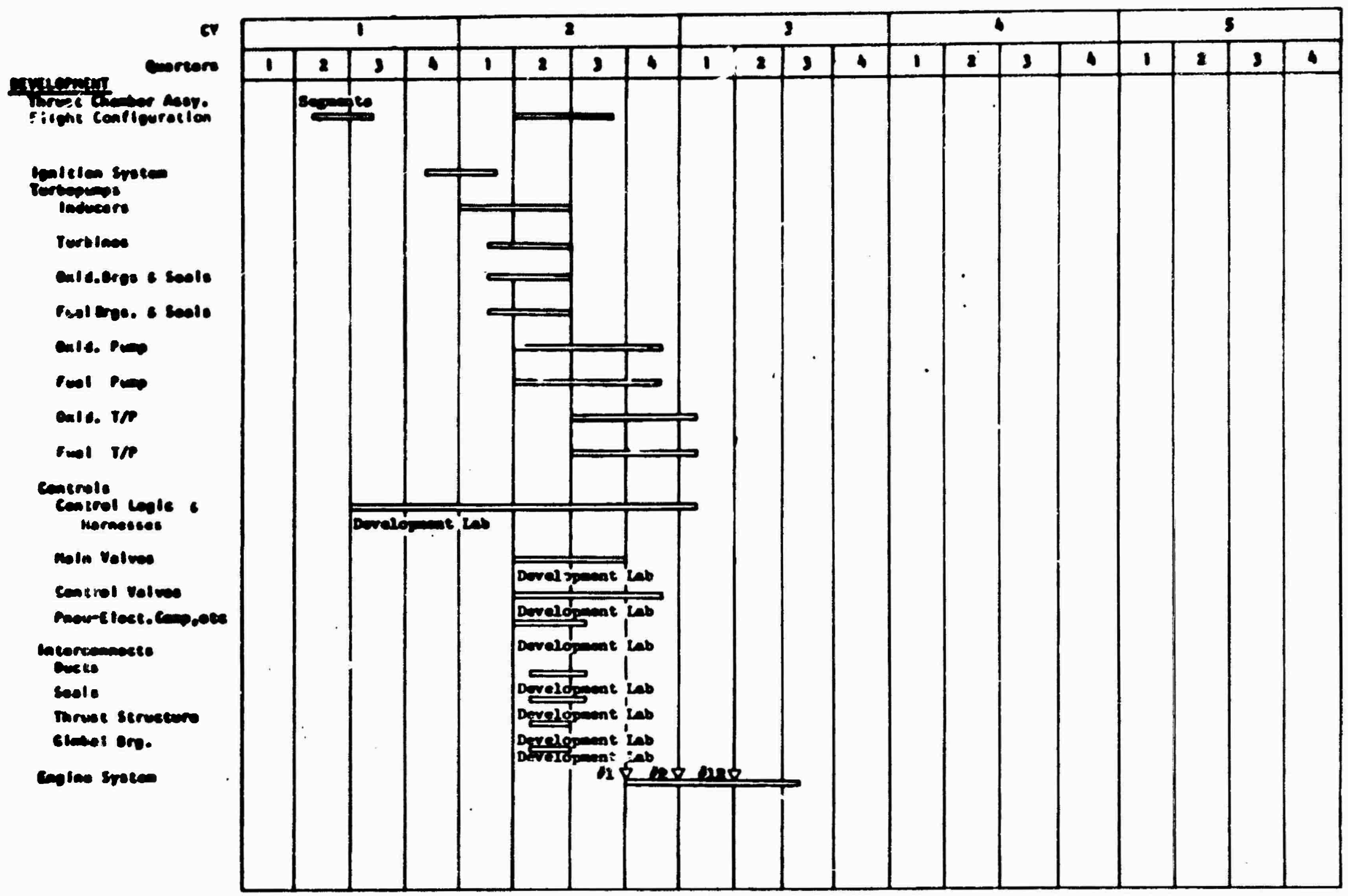


Figure 119. Program Test Plan - Demonstrator Engine
(Aerospike Engine)

Because of the limited effort involved in the demonstrator engine development program, the major impact of ground-based versus space-based maintenance or reusable versus expendible would be in the specific design features incorporated and, therefore, would result in only minor variations in the development effort. Thus, the overall requirements of the space-based demonstrator engine development effort are as previously presented (Fig.119).

ENGINE AND DEVELOPMENT PROGRAM COSTS

The complete engineering development program entails development through flight certification achieving engine readiness for production and field operations. All cost necessary to complete this program were included in the cost estimates and are summarized in the following cost categories:

- Engine and Component Hardware
- Engineering
- Test Consumables
- Test Operations
- Ground Support Equipment
- Tooling and Special Test Equipment
- Manufacturing and Test Facilities

The demonstrator engine program costs represent a program for demonstrating feasibility of the engine concept and establishing a basis for assessing the service life capabilities of the engine design. The demonstrator program includes the same cost categories as the development program except for deletion of ground support equipment development. The first production unit cost includes the cost of fabricating and assembling the engine and the engine checkout test consumables and operation costs.

The primary factors influencing costs are summarized in Table 51. These factors were considered and the costs evaluated, using established cost-scaling relationships, information from previous development program and component cost estimates, and cost experience in previous programs. The aerospike development program plans provided the hardware and test requirements.

TABLE 51. PRIMARY COST FACTORS

Hardware Cost	= f (Hardware Complexity, Hardware Weight, Number of Hardware Units Required)
Engineering Cost	= f (Total Hardware Cost, Number of Engine System Tests)
Consumables Cost	= f (Engine Thrust, Engine Mixture Ratio, Number and Type Tests Required)
Test Operations Costs	= f (Engine Thrust, Number and Type Tests Required)
GSE, Tooling, and STE Cost	= f (Hardware Unit Cost)
Facilities	= f (Engine Thrust, Type Hardware Requiring Test Facilities)
First Unit Cost	= f (Components Required, Component Complexity, Component Weight, Engine Test Cost)

A summary comparison of the cost estimates for the single-panel and double-panel 25,000-pound-thrust expander cycle aerospike engines is presented in Table 52. Costs as presented here are the costs of the actual work required. In establishing a contract price, fee must be added to these costs. The double-panel engine is 2.6 percent more costly to develop and has an 8.4-percent higher production cost than the single-panel engine. Part of this cost difference (about 55 percent for the development program and 50 percent for the first unit cost) is due to the difference in engine operating parameters (P_c and ε), while the remainder results from the different nozzle materials and design differences.

TABLE 52. AEROSPIKE ENGINE COST SUMMARY

(25,000-Pound-Thr

Expander Cy

	P_c	ε	Development Cost, dollars	First Unit Production Cost	Demonstration Program Cost
Reusable Single Panel	750	110	102.0M	761K	27.4M
Resuable Double Panel	1000	200	104.7M	825K	28.2M
Expendable Single Panel	750	110	89.4M	-	-
Expendable Double Panel	1000	200	91.9M	-	-

The double-panel thrust chamber fabrication technique increases the cost of fabricating a combustor segment approximately 15 percent due to the added cost of making and assembling the oxidizer coolant tubes. The oxidizer coolant circuit and a more complex fuel coolant circuit for the double-panel system result in slightly higher pump discharge pressure requirements. This effect increases the cost of fabricating the turbomachinery. Typically, a double-panel engine will cost about 4 percent more than a single-panel engine with the same thrust, chamber pressure, and area ratio, with about 80 percent of the difference due to the thrust chamber assembly and the remainder due to the turbomachinery.

The difference in first unit cost and development program hardware cost is principally due to the difference in the thrust chamber assembly cost. Hardware cost is the largest single factor in the development program cost differential.

The development program cost distribution is illustrated in Fig. 120 for the double-panel aerospike engine. The single-panel engine cost distribution would look very similar, with a slightly smaller portion for the hardware cost.

The unit cost is expected to decrease during production along a 95-percent learning curve. This effect on the double-panel aerospike engine unit cost is shown in Fig. 121, and the single-panel engine unit cost is shown in Fig. 122. For the production rates expected on an engine of this type (one or two engines per month), the development program tooling and special test equipment can be utilized, so new tooling is not required for production.

Applications considered for the aerospike engines require high reliability and long life with minimum operational cost. Achievement of these goals requires control of the engine start, mainstage, operational variations, and shutdown, and is provided by an avionics package. An avionics package provides the elements for sensing and monitoring engine performance, the electronic controller for computing electronic commands to modulate and sequence the valves, and the electrical harnesses to interconnect the components. The avionics sensors provide engine data for performance control, engine readiness checkout, limit control monitoring, and maintenance recording and condition checkout monitoring.

The avionics system incorporates redundant elements to provide a fail-operational/fail-safe capability. No electronic single-point failure will result in loss of avionic function, engine shutdown, or an unsafe operating condition. Low-cost operation of the engine in terms of maintenance actions is ensured by (1) the performance of automated on-board checkout to identify parts for replacement, (2) the verification of engine conditions by monitoring performance in flight, and (3) the acquisition of maintenance data for vehicle recording and ground processing for maintenance trend analysis and preventive maintenance determination.

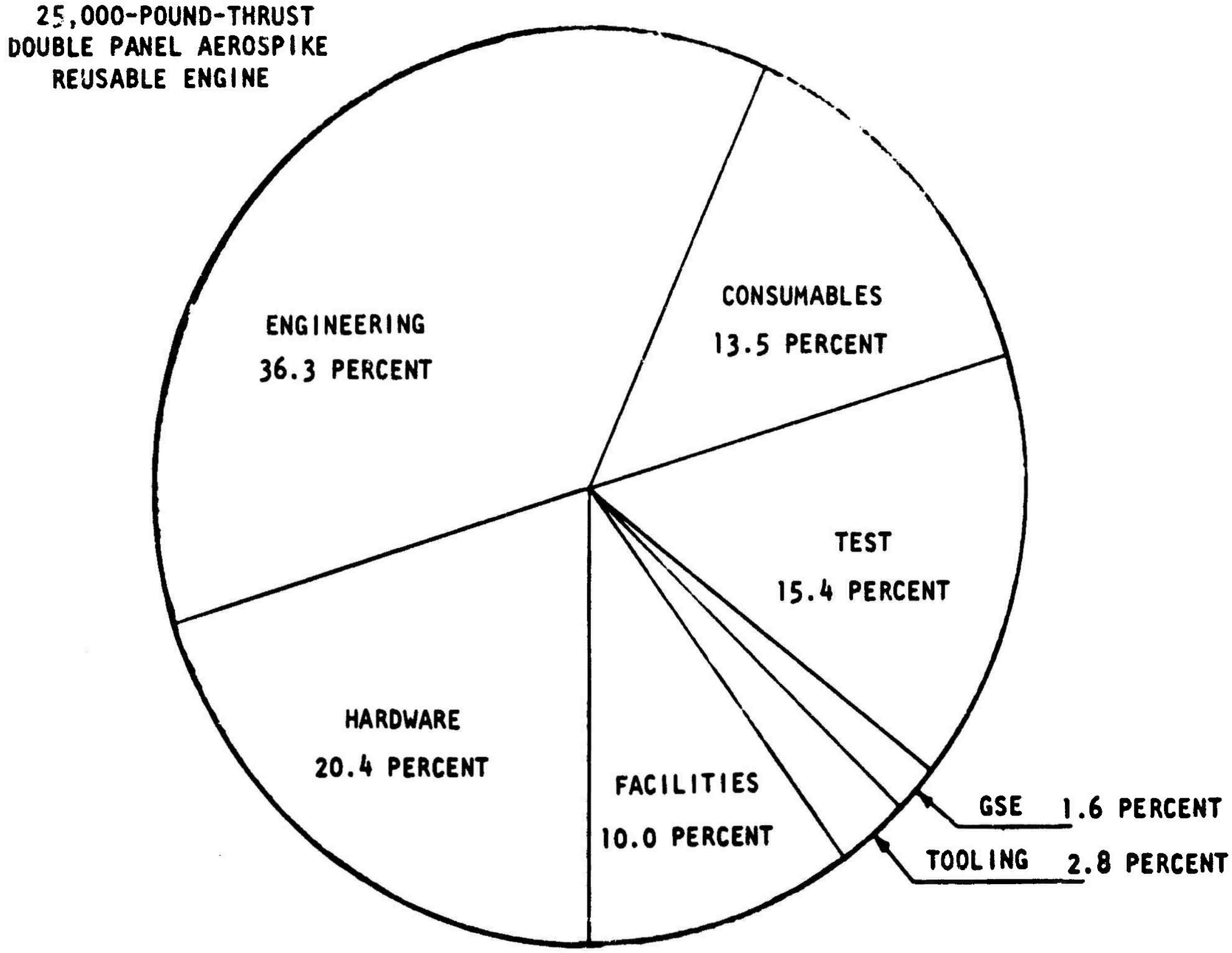


Figure 120. Engineering Development Program Cost Distribution

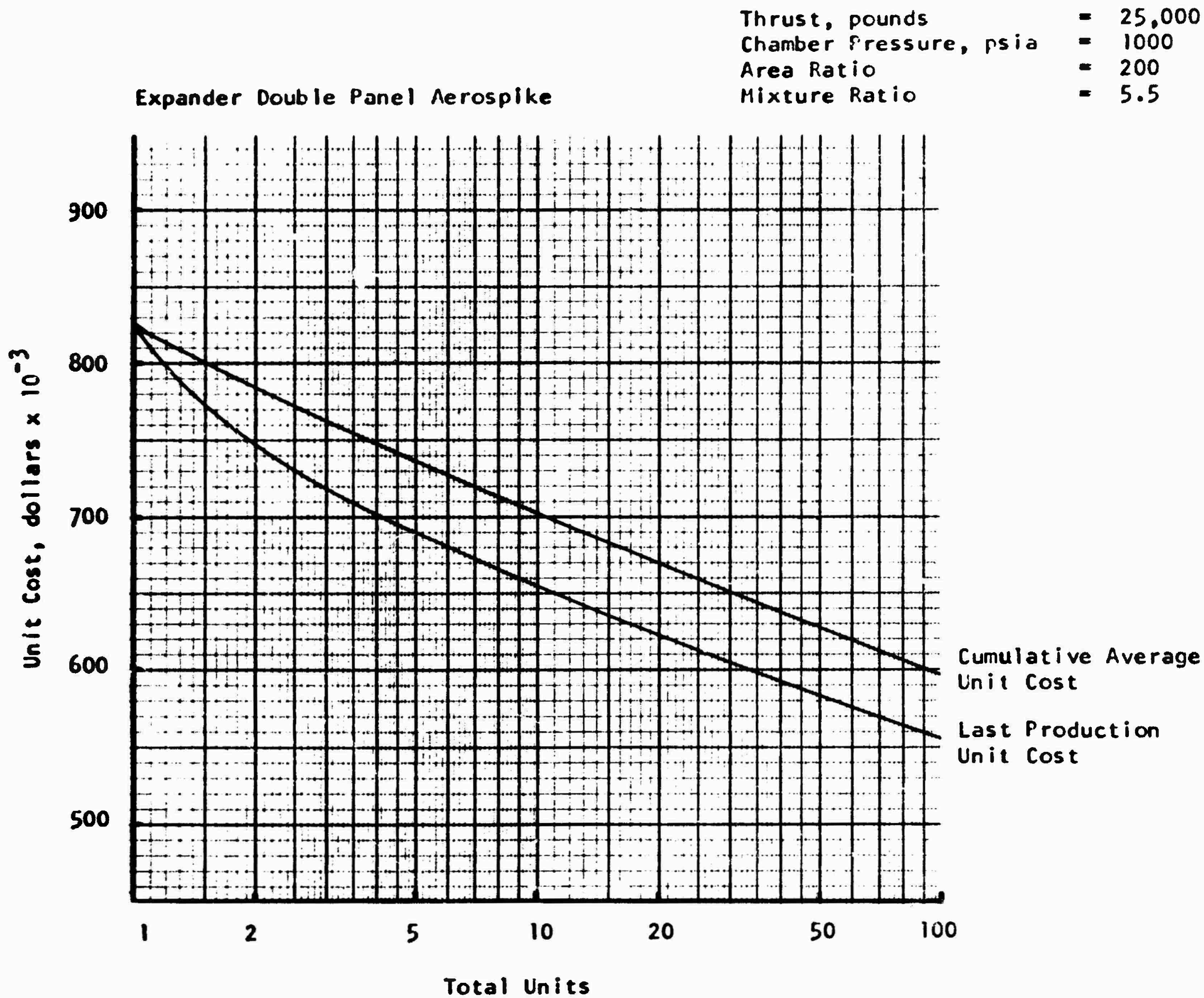


Figure 121. Effect of Learning Curve on Unit Cost (Double Panel)

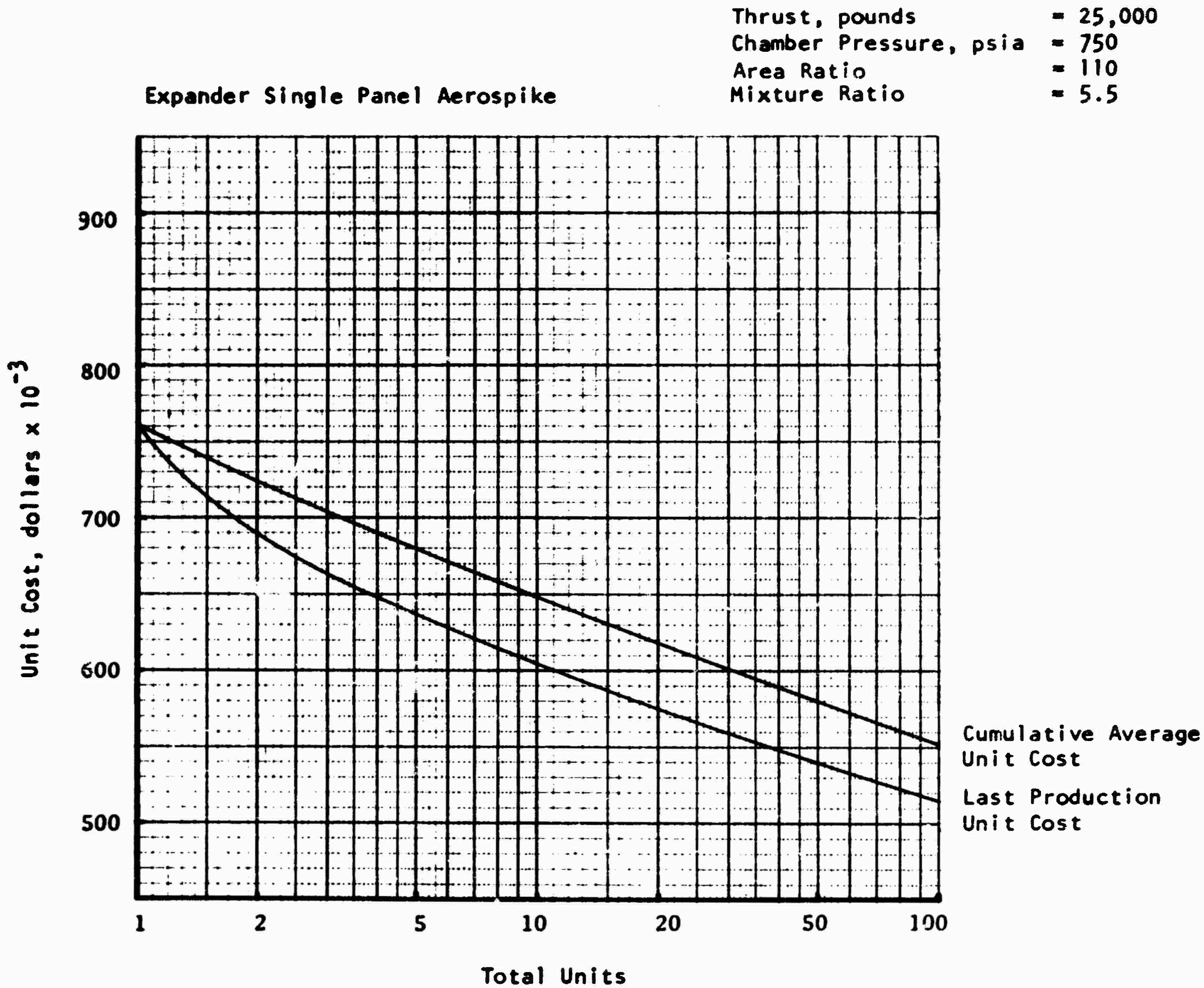


Figure 122. Effect of Learning Curve on Unit Cost (Single Panel)

The avionics system unit cost will depend generally on the need for redundancy, the total number of interfaces (signals, sensors, outputs), and the maintenance control and checkout required. A configuration suitable for the aerospike engine would cost on the order of $600K each. The controller governs the total unit cost, and use of off-the-shelf-type hardware is expected to provide that component at about $400K each. The development costs associated with the avionics package have been estimated to be on the order of $20M to $30M. Because of the possibility that the entire engine avionics package would be provided from a central, vehicle-supplied avionics package, the associated costs were not included in the development program or engine production costs presented in this section.

25,000-POUND-THRUST ENGINE VARYING DESIGN CONDITION ANALYSES

Studies have been conducted to determine how the ground-based, reusable version of the 25,000-pound-thrust aerospike engine is affected by variations in design operating conditions shown in Table 53. The number of thermal cycles for the expendable version also was considered. Each design specification was investigated independently and its impact on dependent operating characteristics was assessed, where applicable. These studies were conducted for both the double- and single-panel engine designs. Tables 54 and 55 present summaries of these studies.

TABLE 53. VARIATIONS IN DESIGN SPECIFICATIONS

Design Specification		Baseline Value	Range Investigated
Engine Mixture Ratio		5.5	5.0 to 7.0
Fuel Pump NPSH, feet		60	0, 15 to 60
Oxidizer Pump NPSH, feet		16	0, 2 to 16
Number of Thermal Cycles:	Reusable	300	60, 600
	Expendable	6	10
Number of Vacuum Starts		60	20, 600
Life, hours		10	2, 20
Maximum Run Time, seconds		1000	500, 2000
Maximum Orbit Storage Time, weeks		52	2, 104
Gimbal Angle, degrees		5	3, 7
Gimbal Acceleration, rad/sec^2		5	10
Nozzle Area Ratio		110 and 200	100 to 400
Throttle Capability		5:1	None, 10:1
Idle Mode Capability		No	Yes

TABLE 54. SUMMARY OF ENGINE VARYING

Independent Engine Design Specification	New Design Value	Engine Specific Impulse, seconds	Engine Weight pounds	Engine Life	Engine Envelope Dimensions, inches	Engine Interface Dimensions	Engine NPSH Requirements	Engine Cooling Requirements
Vacuum Starts	20	No Effect	No Effect				No Effect	No Effect
	600	462.8 (-8.3)	349 (-49)				No Effect	Reduce jacket exit temperature to 756 R (-245)
Nominal Mixture Ratio	5.0:1	474.6 (+3.5)	418 (+20)	No Effect	Dia.=71.9(+4.9) Length=28.8(+1.8)	No Effect		No Effect
	7.0:1	451.3 (-19.8)	342 (-56)	No Effect	Dia.=56.3(-10.7) Length=23.0(-4.0)	No Effect		No Effect
Thermal Cycles	10 (Expendable)	No Effect	No Effect					No Effect
	60 (Reusable)	No Effect	No Effect					No Effect
	600 (Reusable)	462.8 (-8.3)	349 (-49)					Reduce jacket exit temperature to 756 R (-245)
Life	2 hours	No Effect	No Effect					No Effect
	20 hours	No Effect	No Effect					No Effect
Gimbal Angle	3 degrees		No Effect		No Effect	No Effect		
	7 degrees		No Effect		No Effect	No Effect		
Gimbal Acceleration	10 rad/sec^2		408 (+10)		No Effect	No Effect		
Nozzle Area Ratio	100:1	459.4 (-11.7)	318 (-80)	Increased Cycle Life 700 cycles	Dia.=51.0(-16.0) Length=20.8(-6.2)			Reduce jacket exit temperature to 660 R (-341)
	400:1	Not Applicable	Not Applicable	Not Applicable	Not Applicable			Not Applicable
Multiposition Nozzle	Not Applicable	Not Applicable	Not Applicable	Not Applicable	Not Applicable			Not Applicable
H_2 Pump NPSH	0 feet		407 (+9)	No Effect		H_2 inlet dia. increases 0.6 inch		
	15 to 45 feet		407 (+9)	No Effect		H_2 inlet dia. increases 0.6 inch		
	45 to 50 feet		408 (+10)	No Effect		H_2 inlet dia. increases 0.1 inch		
	50 to 60 feet		407 (+9)	No Effect		No Effect		
O_2 Pump NPSH	0 feet		434 (+36)	No Effect		O_2 inlet dia. increases 1.4 inches		
	2 to 8 feet		434 (+36)	No Effect		O_2 inlet dia. increases 1.4 inches		
	8 to 12 feet		423 (+25)	No Effect		O_2 inlet dia. increases 0.2 inch		
	12 to 16 feet		407 (+9)	No Effect		No Effect		
Maximum Run Time	500 seconds		No Effect	No Effect				
	2000 seconds		No Effect	No Effect				
Throttling Capability	None	No effect on nominal value at full thrust	No Effect	No Effect				No Effect
	10:1	No effect on nominal value at full thrust	409 (+11)	No Effect				No Effect
Orbit Storage Time	2 weeks		No Effect	No Effect				
	104 weeks		No Effect	No Effect				
Idle Mode Capability	Low thrust operation under tank head	No effect on nominal value at full thrust, 430 during idle mode	438 (+40)	No Effect		No Effect	No Effect	Reduce engine mixture ratio to 5.9 during idle mode

. SUMMARY OF ENGINE VARYING DESIGN CONDITION ANALYSIS FOR DOUBLE-PANEL ENGINE

Dependent Operating Characteristics										
Engine Interface Dimensions	Engine NPSH Requirements	Engine Cooling Requirements	Engine Maintenance Requirements	Engine Complexity	Gimbal Power Pequirements	TPA Operating Conditions	Technology Improvements Needed	Difference in Demonstrator Engine Program Schedule and Cost	Difference in Development Program Schedule and Cost	Eng Tr
	No Effect	No Effect					No Effect	No Effect	No Effect	
	No Effect	Reduce jacket exit temperature to 756 R (-245)					Improved material thermal cycle fatigue data	$27.8 M (-0.4 M)	$103.6 M (-1.1 M)	
No Effect		No Effect				No Effect	No Effect	$28.4 M (+0.2 M)	$105.4 M (+0.7 M)	
No Effect		No Effect				Increased fuel pump speed	No Effect	$27.7 M (-0.5 M)	$102.9 M (-1.8 M)	
		No Effect	No Effect			No Effect	No Effect	No Effect	No Effect	
		No Effect	No Effect			No Effect	No Effect	No Effect	No Effect	
		Reduce jacket exit temperature to 756 R (-245)	No Effect			Small increase in pump speed	Improved material thermal cycle fatigue data	$27.8 M (-0.4 M)	$103.6 M (-1.1 M)	
		No Effect	No Effect			No Effect	No Effect	$26.0 M (-2.2 M)	$94.5 M (-10.2 M) Schedule reduced 8 months	
		No Effect	No Effect			No Effect	No Effect	$31.0 M (+2.8 M)	$117.0 M (+12.3 M) Schedule increases 10 months	
No Effect					No Effect					
No Effect					No Effect					
No Effect					0.91 HP (+0.12 HP)					
		Reduce jacket exit temperature to 660 R (-341)			0.69 HP (-0.1 HP)		No Effect	$27.7 M (-500 K)	$103.2 M (-1.5 M)	
		Not Applicable			Not Applicable		No Applicable	Not Applicable	Not Applicable	
		Not Applicable	Not Applicable	Not Applicable	Not Applicable		Not Applicable	Not Applicable	Not Applicable	
inlet dia. increases .6 inch				Requires fuel low-pressure pump		Fuel low-pressure pump added	No Effect	$29.1 M (+0.9 M) Schedule increases 0.25 month	$107.9 M (3.2 M) Schedule increases 1.4 months	No
inlet dia. increases .6 inch				Requires fuel low-pressure pump		Fuel low-pressure pump added	No Effect	$28.7 M (0.5 M)	$106.3 M (+1.6 M)	No
inlet dia. increases .1 inch				No Effect		Reduced fuel turbopump speed	No Effect	$28.4 M (+0.2 M)	$105.2 M (+0.5 M)	No
Effect				No Effect		Reduced fuel turbopump speed	No Effect	$28.4 M (+0.2 M)	$105.2 M (+0.5 M)	No
inlet dia. increases .4 inches				Requires oxidizer low-pressure pump		Low-pressure pump added	Pump two-phase oxygen	$29.5 M (+1.3 M) Schedule increases 0.25 month	$109.2 M (+4.5 M) Schedule increases 1.4 months	No
inlet dia. increases .4 inches				Requires oxidizer low-pressure pump		Low-pressure pump added	No Effect	$29.1 M (+0.9 M)	$107.6 M (+2.9 M)	No
inlet dia. increases .2 inch				No Effect		Reduced oxidizer turbopump speed	No Effect	$28.6 M (+0.4 M)	$106.2 M (+1.5 M)	No
Effect				No Effect		Same as above	No Effect	$28.3 M (+0.1 M)	$105. 1 M (+0.4)	No
								No Effect	No Effect	
								No Effect	No Effect	
		No Effect		No Effect			No Effect	$27.4 M (-0.8 M) Schedule reduced 0.5 month	$101.7 M (-3.0 M) Schedule reduced 1.8 months	No
		No Effect		No Effect			No Effect	$29.0 M (+0.8 M) Schedule increases 0.4 month	$107.4 M (2.7 M) Schedule increases 1.2 months	No sta seq alt
				No Effect			Protection against vacuum cold welding	No Effect	No Effect	
				No Effect			Protection against vacuum cold welding	No Effect	No Effect	
Effect	No Effect	Reduce engine mixture ratio to 3.9 during idle mode		Requires fuel turbine inlet valve		Turbopumps are inoperative	No Effect	$29.6 M (+1.4 M) Schedule increases 0.6 month	$109.4 M (+4.7 M) Schedule increases 2.0 months	No sta seq alt

2

ENGINE

Technology Improvements Needed	Difference in Demonstrator Engine Program Schedule and Cost	Difference in Development Program Schedule and Cost	Engine Start Transient	Nozzle Area Ratio	Difference in First Production Unit Cost
No Effect	No Effect	No Effect		No Effect	No Effect
Improved material thermal cycle fatigue data	$27.8 M (-0.4 M)	$10[illegible] M (-1.1 M)		Reduce ε to 126	$795 K (-30 K)
No Effect	$28.4 M ([illegible].2 M)	$105.4 M (+0.7 M)		Increase ε to 232:1	$840 K (+15 K)
No Effect	$27.7 M (-0.5 M)	$102.9 M (-1.8 M)		Reduce ε to 133:1	$790 K (-35 K)
No Effect	No Effect	No Effect		No Effect	No Effect
No Effect	No Effect	No Effect		No Effect	No Effect
Improved material thermal cycle fatigue data	$27.8 M (-0.4 M)	$103.6 M (-1.1 M)		Reduce ε to 126	$795 K (-30 K)
No Effect	$26.0 M (-2.2 M)	$94.5 M (-10.2 M) Schedule reduced 8 months			No Effect
No Effect	$31.0 M (+2.8 M)	$117.0 M (+12.3 M) Schedule increases 10 months			No Effect
					No Effect.
					No Effect
					No Effect
No Effect	$27.7 M (-500 K)	$103.2 M (-1.5 M)			$784 K (-41 K)
No Applicable	Not Applicable	Not Applicable			Not Applicable
Not Applicable	Not Applicable	Not Applicable		Not Applicable	Not Applicable
No Effect	$29.1 M (+0.9 M) Schedule increases 0.25 month	$107.9 M (3.2 M) Schedule increases 1.4 months	No Effect		$846 K (+21 K)
No Effect	$28.7 M (+0.5 M)	$106.3 M (+1.6 M)	No Effect		$846 K (+21 K)
No Effect	$28.4 M (+0.2 M)	$105.2 M (+0.5 M)	No Effect		$835 K (+10 K)
No Effect	$28.4 M (+0.2 M)	$105.2 M (+0.5 M)	No Effect		$835 K (+10 K)
Pump two-phase oxygen	$29.5 M (+1.3 M) Schedule increases 0.25 month	$109.2 M (+4.5 M) Schedule increases 1.4 months	No Effect		$873 K (+48 K)
No Effect	$29.1 M (+0.9 M)	$107.6 M (+2.9 M)	No Effect		$873 K (+48 K)
No Effect	$28.6 M (+0.4 M)	$106.2 M (+1.5 M)	No Effect		$854 K (+29 K)
No Effect	$28.3 M (+0.1 M)	$105. 1 M (+0.4)	No Effect		$833 K (+ 8 K)
	No Effect	No Effect			No Effect
	No Effect	No Effect			No Effect
No Effect	$27.4 M (-0.8 M) Schedule reduced 0.5 month	$101.7 M (-3.0 M) Schedule reduced 1.8 months	No Effect		No Effect
No Effect	$29.[illegible] M (+0.8 M) Schedule increases 0.4 month	$107.4 M (+2.7 M) Schedule increases 1.2 months	No effect on start time, sequence altered		$839 K (+14 K)
Protection against vacuum cold welding	No Effect	No Effect			No Effect
Protection against vacuum cold welding	No Effect	No Effect			No Effect
No Effect	$29.6 M (+1.4 M) Schedule increases 0.6 month	$109.4 M (+4.7 M) Schedule increases 2.0 months	No effect on start time, sequence altered		$855 K (+30 K)

3

TABLE 55. SUMMARY OF ENGINE VARYING DESIGN COND

Independent Engine Design Specification	New Design Valve	Engine Specific Impulse, seconds	Engine Weight, pounds	Engine Life	Engine Envelope Dimensions, inches	Engine Interface Dimensions, inches	Engine NPSH Requirements	Engine Cooling Requiremen
Vacuum Starts	20	No Effect	No Effect				No Effect	No Effect
	600	455.5 (-2.5)	346 (-14)				No Effect	Reduce Maxi Hot-Gas Wal Temperature _ R
Nominal Mixture Ratio	5.0:1	460.0 (+2.0)	365 (+5)	No Effect	No Effect	No Effect		Jacket Exit Temperature Reduced to 780 R (-60)
	7.0:1	442.8 (-15.2)	340 (-20)	No Effect	Diameter = 59 (-3) Length = 23 (-1)	No Effect		Jacket Exit Temperature Increased t 970 R (+130
Thermal Cycles	10 (Expendable)	No Effect	No Effect					No Effect
	60 (Reusable)	No Effect	No Effect					No Effect
	600 (Reusable)	455.5 (-2.5)	346 (-14)					Reduce Maxi Hot-Gas Wal Temperature by 65 R
Life	2 Hours	No Effect	No Effect					No Effect
	20 Hours	No Effect	No Effect					No Effect
Gimbal Angle	3 Degrees		No Effect		No Effect	No Effect		
	7 Degrees		No Effect		No Effect	No Effect		
Gimbal Acceleration	10 rad/sec^2		370 (+10)	No Effect		No Effect		
Nozzle Area Ratio	100:1	456.4 (-1.6)	360 (-10)	Not Applicable	Diameter = 59 (-3) Length = 23 (-1)			Jacket Exi Temperatur Decreased 795 R (-45
	400:1	Not Applicable	Not Applicable	Not Applicable	Not Applicable			Not Applic
Multiposition Nozzle	Not Applicable	Not Applicable	Not Applicable	Not Applicable	Not Applicable			Not Applic
H_2 Pump NPSH	0 Feet		362 (+2)	No Effect		Fuel Inlet = 3.1 (+0.6)		
	15 to 20 Feet		362 (+2)	No Effect		Fuel Inlet = 3.1 (+0.6)		
	20 to 25 Feet		381 (+21)	No Effect		Fuel Inlet = 2.9 (+0.4)		
	25 to 60 Feet		378 (+18)	No Effect		Fuel Inlet = 2.8 (+0.3)		
O_2 Pump NPSH	0 Feet		386 (+26)	No Effect		Oxidizer Inlet = 4.4 (+1.4)		
	2 to 4 Feet		386 (+26)	No Effect		Oxidizer Inlet = 4.4 (+1.4)		
	4 to 16 Feet		396 (+36)	No Effect		Oxidizer Inlet = 3.5 (+0.5)		
Maximum Run Time	500 Seconds		No Effect	No Effect				
	2000 Seconds		No Effect	No Effect				
Throttling Capability	None	No Effect on Nominal Value at Full Thrust	No Effect	No Effect				No Effect
	10:1	No Effect on Nominal Value at Full Thrust	371 (+11)	No Effect				No Effect
Orbit Storage Time	2 Weeks		No Effect	No Effect				
	104 Weeks		No Effect	No Effect				
Idle Mode Capability	Low-Thrust Operation Under Tank Head	No Effect on Nominal Value at Full Thrust, 415 During Idle Mode	400 (+40)	No Effect		No Effect	No Effect	Reduce M Ratio to During Id

MMARY OF ENGINE VARYING DESIGN CONDITION ANALYSIS FOR SINGLE-PANEL ENGINE

Dependent Operating Characteristics										
Engine Envelope Dimensions, inches	Engine Interface Dimensions, inches	Engine NPSH Requirements	Engine Cooling Requirements	Engine Maintenance Requirements	Complexity	Gimbal Power Requirements	TPA Operating Conditions	Technology Improvements Needed	Difference in Demonstrator Engine Program Schedule and Costs	Difference in Development Program Schedule and Costs
		No Effect	No Effect					No Effect	No Effect	No Effect
		No Effect	Reduce Maximum Hot-Gas Wall Temperature by 65 R					Improved Material Thermal Cycle Fatigue Data	$27.3 M (-0.1 M)	$101.8 M (-0.2 M)
No Effect	No Effect		Jacket Exit Temperature Reduced to 780 R (-60)				Slightly Higher Turbopump Speed	No Effect	$27.5 M (+0.1 M)	$102.3M(+0.3 M)
Diameter = 59 (-3) Length = 23 (-1)	No Effect		Jacket Exit Temperature Increased to 970 R (+130)				Slightly Higher Turbopump Speed	No Effect	$27.1 M (-0.3 M)	$101.0M(-1.0 M)
			No Effect	No Effect			No Effect	No Effect	No Effect	No Effect
			No Effect	No Effect			No Effect	No Effect	No Effect	No Effect
			Reduce Maximum Hot-Gas Wall Temperature by 65 R	No Effect			Increase Fuel Turbopump Speed by 6.1 Percent	Improved Material Thermal Cycle Fatigue Data	$27.3 M (-0.1 M)	$101.8 M (-0.2 M)
			No Effect	No Effect			No Effect	No Effect	$25.0 M (-2.4 M)	$92.0 M (-10.0 M) Schedule Reduced 8 Months
			No Effect	No Effect			No Effect	No Effect	$30.0 M (+2.6 M)	$114.5 M (+12.5 Schedule Increas 10 Months
No Effect	No Effect					No Effect				
No Effect	No Effect					No Effect				
	No Effect			No Effect		0.8 hp (+0.1 hp)				
Diameter = 59 (-3) Length = 23 (-1)			Jacket Exit Temperature Decreased to 795 R (-45)			0.69 hp (-0.01 hp)		No Effect	$27.34 M (-0.06 M)	$101.8 M (-0.2 M
Not Applicable			Not Applicable			Not Applicable		Not Applicable	Not Applicable	Not Applicable
Not Applicable			Not Applicable	Not Applicable	Not Applicable	Not Applicable		Not Applicable	Not Applicable	Not Applicable
	Fuel Inlet = 3.1 (+0.6)				Fuel Low-Pressure Pump Required		Fuel Low-Pressure Pump Added	No Effect	$28.2 M (+0.8 M) Schedule Increases 0.25 Months	$105.2 M (+3.2 M Schedule Increases 1.4 Months
	Fuel Inlet = 3.1 (+0.6)				Fuel Low-Pressure Pump Required		Fuel Low-Pressure Pump Added	No Effect	$27.8 M (+0.4 M)	$103.6 M (+1.6 M
	Fuel Inlet = 2.9 (+0.4)				No Effect		Reduced Fuel Turbopump Speed, Turbine Bypass Flow Reduced to 15 Percent	No Effect	$27.8 M (+0.4 M)	$103.5 M (+1.5 M
	Fuel Inlet = 2.8 (+0.3)				No Effect		Reduced Fuel Turbopump Speed	No Effect	$27.7 M (+0.3 M)	$103.3 M (+1.3
	Oxidizer Inlet = 4.4 (+1.4)				Oxidizer Low-Pressure Pump Required		Oxidizer Low-Pressure Pump Added	Pumping Two-Phase Oxygen	$28.8 M (+1.4 M) Schedule Increases 0.25 Month	$107.1 M (+5.1 Schedule Increases 1.4 Months
	Oxidizer Inlet = 4.4 (+1.4)				Oxidizer Low-Pressure Pump Required		Oxidizer Low-Pressure Pump Added	No Effect	$28.4 M (+1.0 M)	$105.4 M (+3.4
	Oxidizer Inlet = 3.5 (+0.5)				No Effect		Reduced Oxidizer Turbopump Speed	No Effect	$28.1 M (+0.7 M)	$104.5 M (+2.5
									No Effect	No Effect
									No Effect	No Effect
			No Effect		No Effect			No Effect	$26.6 M (-0.8 M) Schedule Reduced 0.5 Month	$99.0 M (-3.0 M Schedule Reduce 1.8 Months
			No Effect		No Effect			No Effect	$28.2 M (+0.8 M) Schedule Increases 0.4 Month	$104.7 M (+2.7 Schedule Increases 1.2 Months
					No Effect			Protection Against Vacuum Cold Welding	No Effect	No Effect
					No Effect			Protection Against Vacuum Cold Welding	No Effect	No Effect
	No Effect	No Effect	Reduce Mixture Ratio to 3.7 During Idle Mode		Fuel Turbine Inlet Valve Required		Turbopumps Inoperative During Idle Mode	No Effect	$28.8 M (+1.4 M) Schedule Increases 0.6 Month	$106.7 M (+4.7 Schedule Increases 2.0 Months

2

	Technology Improvements Needed	Difference in Demonstrator Engine Program Schedule and Costs	Difference in Development Program Schedule and Costs	Engine Start Transient	Nozzle Area Ratio	Difference in First Production Unit Cost
	No Effect	No Effect	No Effect		No Effect	No Effect
	Improved Material Thermal Cycle Fatigue Data	$27.3 M (-0.1 M)	$101.8 M (-0.2 M)		Reduce ε to 95	$754 K (-7 K)
her eed	No Effect	$27.5 M (+0.1 M)	$102.3M(+0.3 M)		No Effect	$765 K (+4 K)
her eed	No Effect	$27.1 M (-0.3 M)	$101.0M(-1.0 M)		Reduce ε to 100	$750 K (-9 K)
	No Effect	No Effect	No Effect		No Effect	No Effect
	No Effect	No Effect	No Effect		No Effect	No Effect
eed t	Improved Material Thermal Cycle Fatigue Data	$27.3 M (-0.1 M)	$101.8 M (-0.2 M)		Reduce ε to 95	$754 K (-7 K)
	No Effect	$25.0 M (-2.4 M)	$92.0 M (-10.0 M) Schedule Reduced 8 Months			No Effect
	No Effect	$30.0 M (+2.6 M)	$114.5 M (+12.5 M) Schedule Increased 10 Months			No Effect
						No Effect
						No Effect
						No Effect
	No Effect	$27.34 M (-0.06 M)	$101.8 M (-0.2 M)			$756 K (-5 K)
	Not Applicable	Not Applicable	Not Applicable			Not Applicable
	Not Applicable	Not Applicable	Not Applicable		Not Applicable	Not Applicable
p	No Effect	$28.2 M (+0.8 M) Schedule Increases 0.25 Months	$105.2 M (+3.2 M) Schedule Increases 1.4 Months	No Effect		$783 K (+22 K)
p	No Effect	$27.8 M (+0.4 M)	$103.6 M (+1.6 M)	No Effect		$783 K (+22 K)
eed, ss to	No Effect	$27.8 M (+0.4 M)	$103.5 M (+1.5 M)	No Effect		$790 K (+29 K)
eed	No Effect	$27.7 M (+0.3 M)	$103.3 M (+1.3 M)	No Effect		$786 K (+25 K)
- p	Pumping Two-Phase Oxygen	$28.8 M (+1.4 M) Schedule Increases 0.25 Month	$107.1 M (+5.1 M) Schedule Increases 1.4 Months	No Effect		$820 K (+59 K)
- p	No Effect	$28.4 M (+1.0 M)	$105.4 M (+3.4 M)	No Effect		$820 K (+59 K)
izer eed	No Effect	$28.1 M (+0.7 M)	$104.5 M (+2.5 M)	No Effect		$810 K (+49 K)
		No Effect	No Effect			No Effect
		No Effect	No Effect			No Effect
	No Effect	$26.6 M (-0.8 M) Schedule Reduced 0.5 Month	$99.0 M (-3.0 M) Schedule Reduced 1.8 Months	No Effect		$761 K (0.0)
	No Effect	$28.2 M (+0.8 M) Schedule Increases [illegible] Month	$104.7 M (+2.7 M) Schedule Increases 1.2 Months	No Effect On Time, Sequence Altered		$775 K (+14 K)
	Protection Against Vacuum Cold Welding	No Effect	No Effect			No Effect
	Protection Against Vacuum Cold Welding	No Effect	No Effect			No Effect
Mode	No Effect	$28.8 M (+1.4 M) Schedule Increases 0.6 Month	$106.7 M (+4.7 M) Schedule Increases 2.0 Months	No Effect On Time, Sequence Altered		$791 K (+30 K)

3

BASELINE ENGINE DESCRIPTIONS

The baseline configurations used in these studies were the demonstrator single-panel (P_c = 750, ε = 110) and the optimum double-panel (P_c = 1000, ε = 200) aerospike engines designed for a thrust of 25,000 pounds. Both engines operate on an expander topping cycle. This cycle utilizes the major portion of the fuel, after it passes through the thrust chamber regenerative cooling jacket, as an energy source for the axial flow impulse turbines. The turbines, in a parallel flow arrangement, directly drive centrifugal pumps in the propellant feed system. Nearly all of the turbine exhaust and turbine bypass flows are injected and combusted with the oxidizer in the thrust chamber. A small portion of the hot-gas fuel flow is used as base bleed. The single-panel engine injector utilizes liquid oxygen, but with the double-panel design the oxidizer also is heated in a regenerative jacket before it flows through the injector elements.

Both engine control systems consist of main propellant isolation valves immediately upstream of the pumps, an oxidizer turbine inlet valve controls engine mixture ratio by altering the turbine flow split. The base flow is controlled with an orifice.

The engines are designed for multiple altitude starts and utilize a tank-head start sequence. Fuel flows through the regenerative jacket under tank pressure where it absorbs residual heat. The warm hydrogen is then expanded through the turbines, according to a controlled sequence, and provides power to the pumps. After ignition, the engine bootstraps to full thrust.

The combustion chamber is constructed of individual segment liners fabricated from cast NARLoy with brazed-on coolant channel closeout panels. Two continuous titanium alloy rings provide structural support to the segmented chamber. The nozzle utilizes a tubular wall-type construction. The individual tubes are tapered and provide a single-pass cooling circuit in the nozzle.

A summary of the basic engine system designs and operating parameters is presented in Table 56. Descriptions of the turbopumps are shown in Table 57.

TABLE 56. ENGINE SYSTEM DESIGN AND OPERATING PARAMETERS

Cooling Circuit	Double-Panel	Single-Panel
Thrust, pounds	25,000	25,000
Chamber Pressure, psia	1000	750
Nozzle Expansion Area Ratio	200	110
Nozzle Percent Length	20	20
Engine Mixture Ratio	5.5	5.5
Thrust Chamber Mixture Ratio	5.57	5.57
Specific Impulse, sec	471	458
Base Flowrate, lb/sec	0.11	0.11
Hydrogen Injection Temperature, R	938	804
Turbine Inlet Temperature, R	1001	840
Percent Turbine Bypass	12	20
Engine Weight, pounds	398	360
Engine Length, inches	27	24
Engine Diameter, inches	67	62

TABLE 57. TURBOPUMP DESCRIPTIONS

Cooling Circuit	Double-Panel		Single-Panel	
Propellant	Fuel	Oxidizer	Fuel	Oxidizer
Pump				
Number of Stages	3	1	2	1
Flowrate, lb/sec	8.16	44.9	8.4	46.2
Pump Pressure Rise, psi	3255	1711	1560	1032
Pump Speed, rpm	75,000	22,000	75,000	22,000
Efficiency	0.620	0.665	0.657	0.680
Pump Horsepower	2320	420	1150	256
Impeller Tip Diameter, inches	4.3	4.9	3.90	3.81
Stage Specific Speed, rpm-gpm$^{0.5}$/ft$^{0.75}$	870	825	1105	1230
Pump NPSH, feet	60	16	60	16
Turbine				
Number of Rows	1	1	1	1
Flowrate, lb/sec	5.05	2.24	4.90	1.77
Inlet Temperature, R	1001	1001	840	840
Pressure Ratio	1.66	1.43	1.3	1.24
Efficiency	0.687	0.39	0.786	0.597
Pitch Diameter, inches	5.0	8.0	5.0	8.0
Turbine DN, mm-rpm	1.9×10^6	0.6×10^6	1.9×10^6	0.6×10^6
Percent Admission	100	20	100	20
Turbine $A_A N^2$, in.^{2}-rpm^2	27×10^9	4.7×10^9	55×10^9	6.5×10^9
Turbine U/C	0.34	0.19	0.50	0.26
Turbine Tip Speed, ft/sec	1743	804	1841	819

ANALYSES OF DOUBLE-PANEL ENGINE

VARIATION OF DESIGN ENGINE MIXTURE RATIO

The baseline engine system has a design mixture ratio of 5.5. The effects of varying this design specification over a range from 5 to 7 are summarized.

Nozzle Area Ratio

The chamber pressure (1000 psia) and nozzle area ratio (200) of the baseline engine were chosen to provide optimum mission potential using a trade factor of 42.6 pounds of engine weight per second of delivered specific impulse. Optimization of mission potential and, therefore, chamber pressure and area ratio, is dependent on thrust chamber and nozzle cooling requirements which are established by duration and cycle life considerations. As the design mixture ratio is increased, the nozzle area ratio must be decreased to maintain the specified life requirements. The effect on nozzle area ratio is shown in Fig. 123.

Specific Impulse

The effect of varying mixture ratio and the associated change in nozzle area ratio on delivered specific impulse is shown in Fig. 124 . Engine performance decreases as mixture ratio increases.

Weight

The effect of varying mixture ratio on engine system dry weight is shown in Fig. 125 . Engine weight decreases as mixture ratio increases, principally due to the effect of nozzle area ratio on engine diameter.

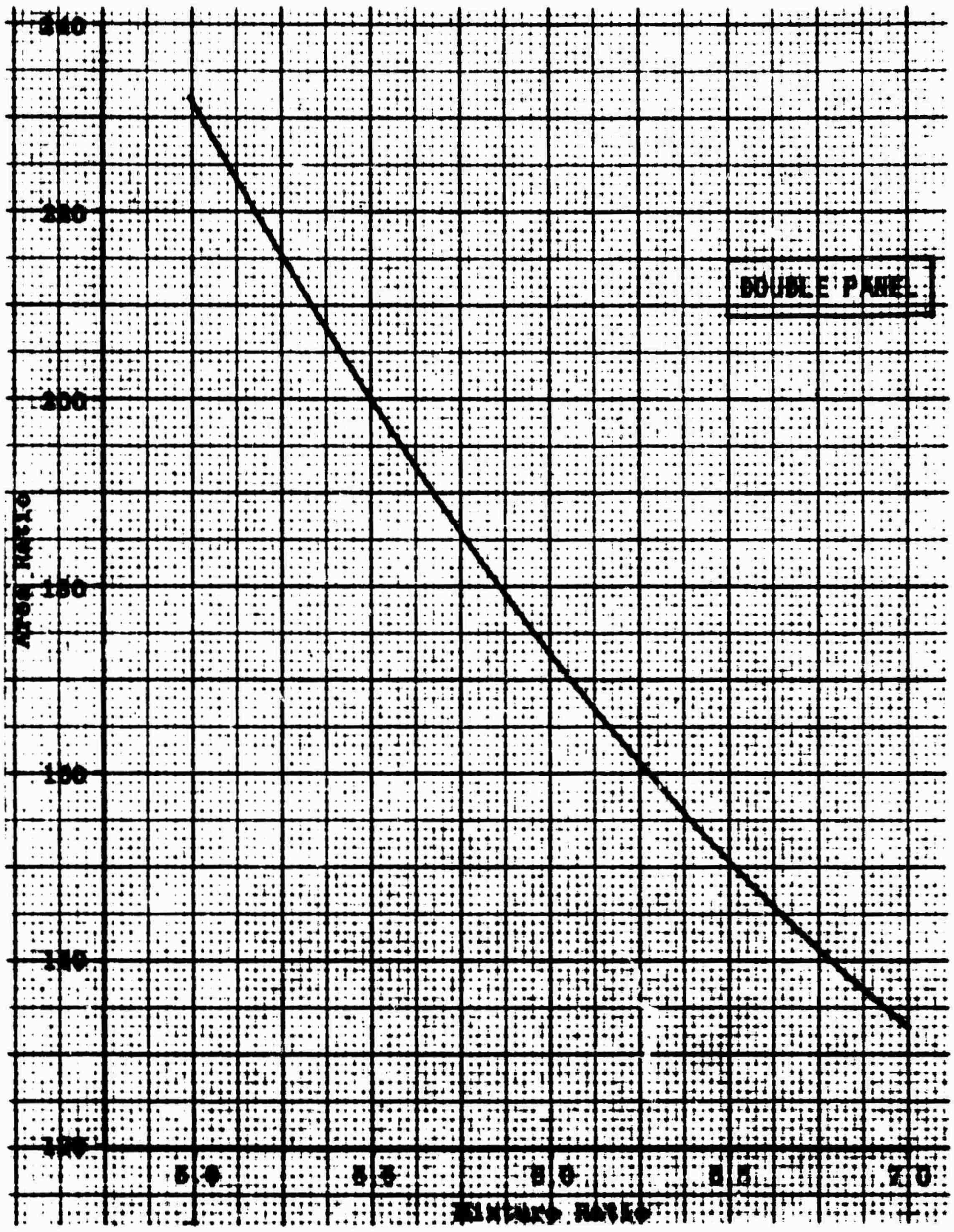


Figure 123. Effect of Design Engine Mixture Ratio on Nozzle Area Ratio

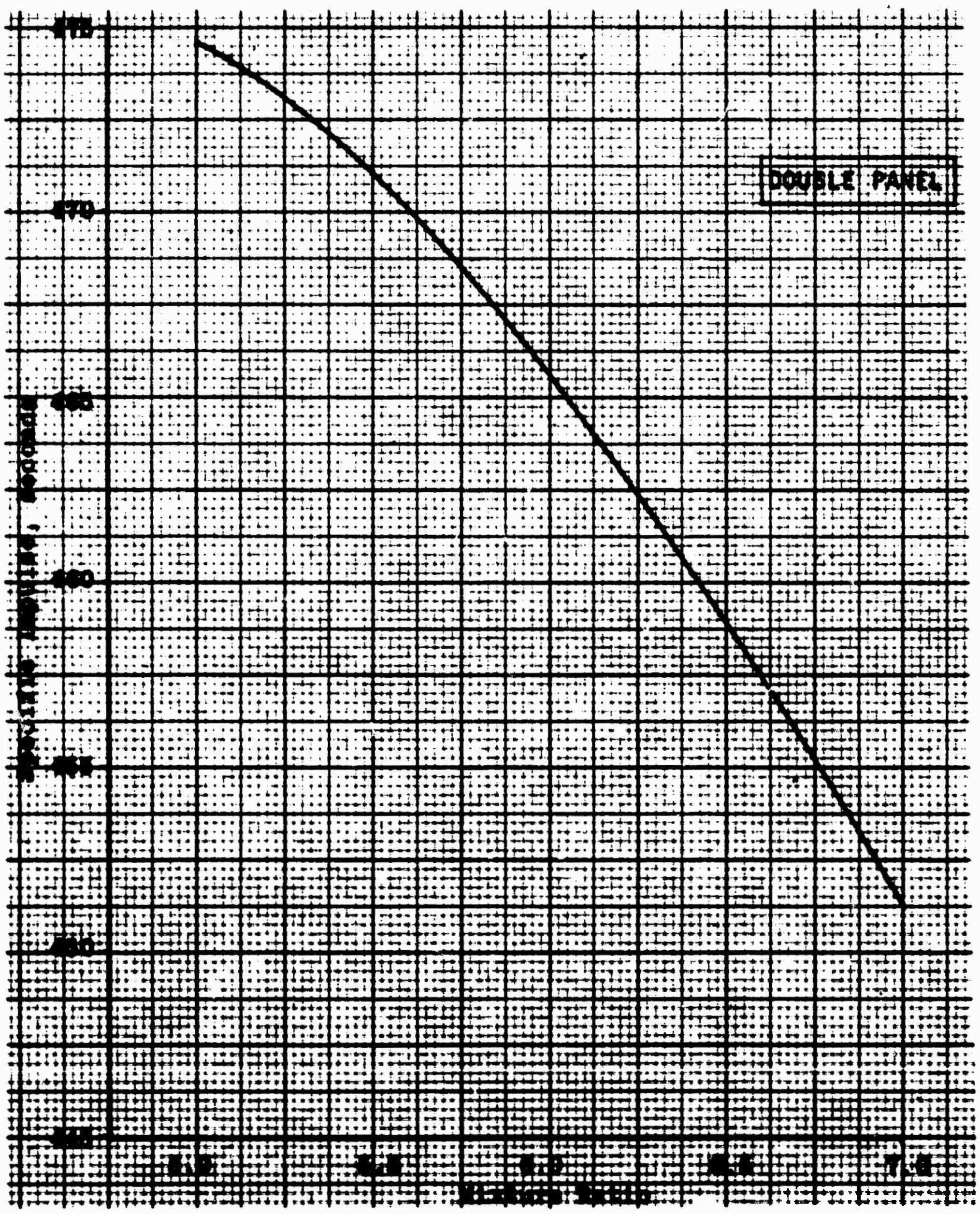


Figure 124. Effect of Design Engine Mixture Ratio on Delivered Vacuum Engine Specific Impulse

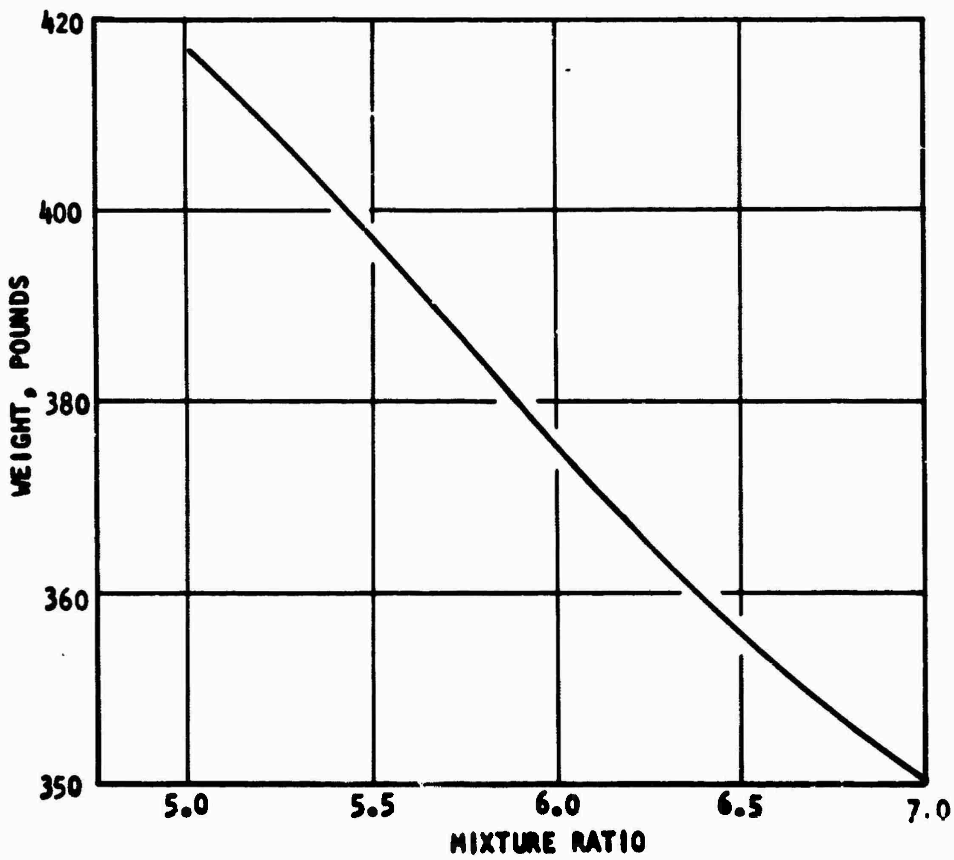


Figure 125. Effect of Design Engine Mixture Ratio on Double-Panel Engine System Dry Weight

Envelope Dimensions

The effects of varying mixture ratio on engine envelope dimensions are shown in Fig. 126. The variations in length and diameter are due to changes in the nozzle area ratio.

Interface Dimensions

Variation of engine mixture ratio over a range from 5 to 7 does not affect dimensions. The inside diameter of the fuel inlet duct is 2.5 inches and the oxidizer duct diameter is 3.0 inches.

Cooling Requirements

Thrust chamber cooling requirements are not affected by variations in the design mixture ratio due to the ground rules of this study. The nozzle area ratio was varied to maintain a constant maximum hot-gas wall temperature and the material temperature gradients do not change significantly for this design criteria.

Engine System Operating Conditions

The effects of varying the design mixture ratio on turbomachinery parameters are shown in Fig. 127. The turbine bypass flow was maintained at 12 percent of the total fuel flow.

Technology Improvements Needed

No technology improvements are needed for variations in the design mixture ratio. Variation of the nozzle area to meet life specifications results in operating conditions which do not exceed the design criteria established for the baseline engine.

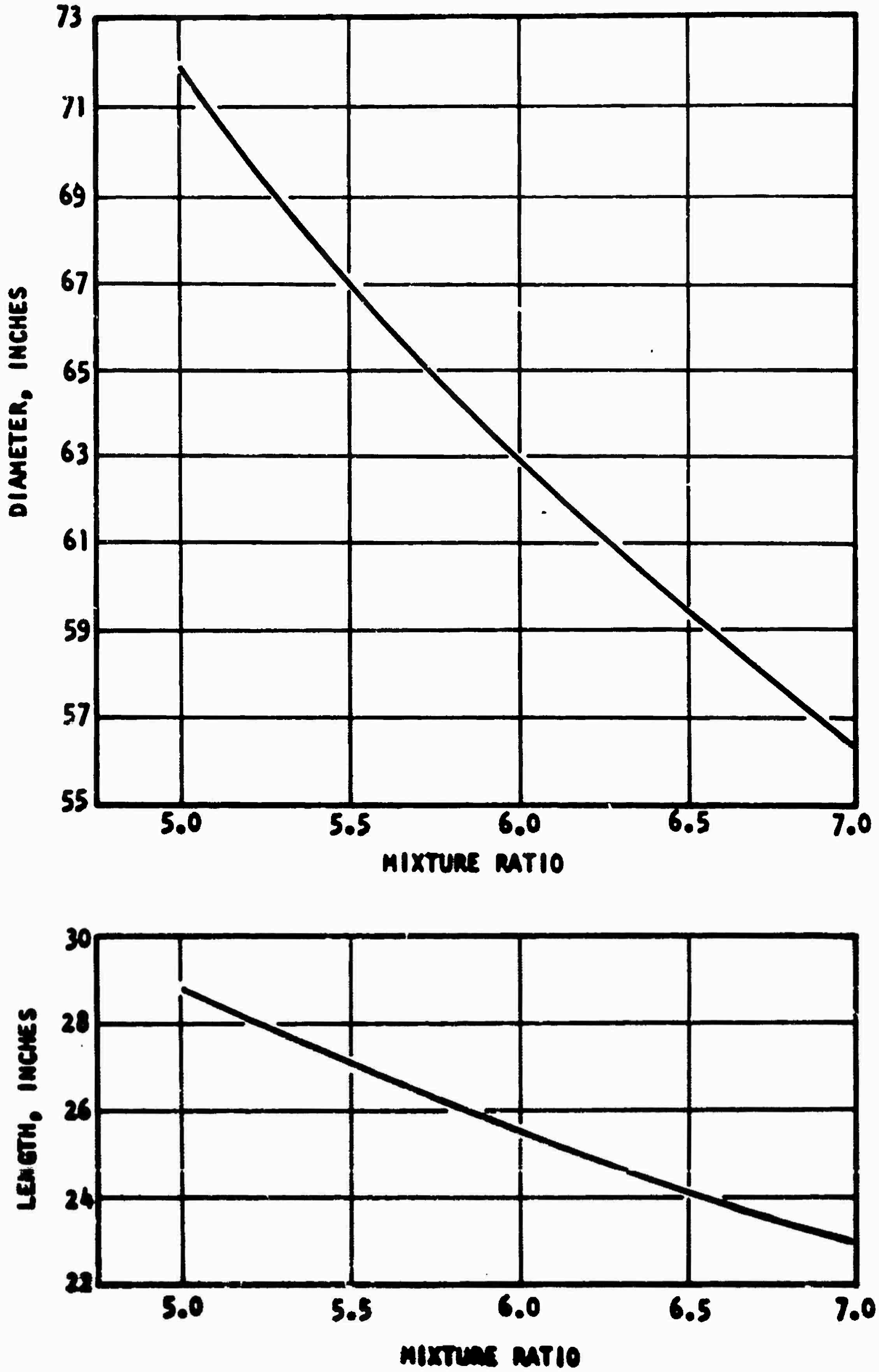


Figure 126. Effect of Design Engine Mixture Ratio on Double-Panel Engine Envelope Dimensions

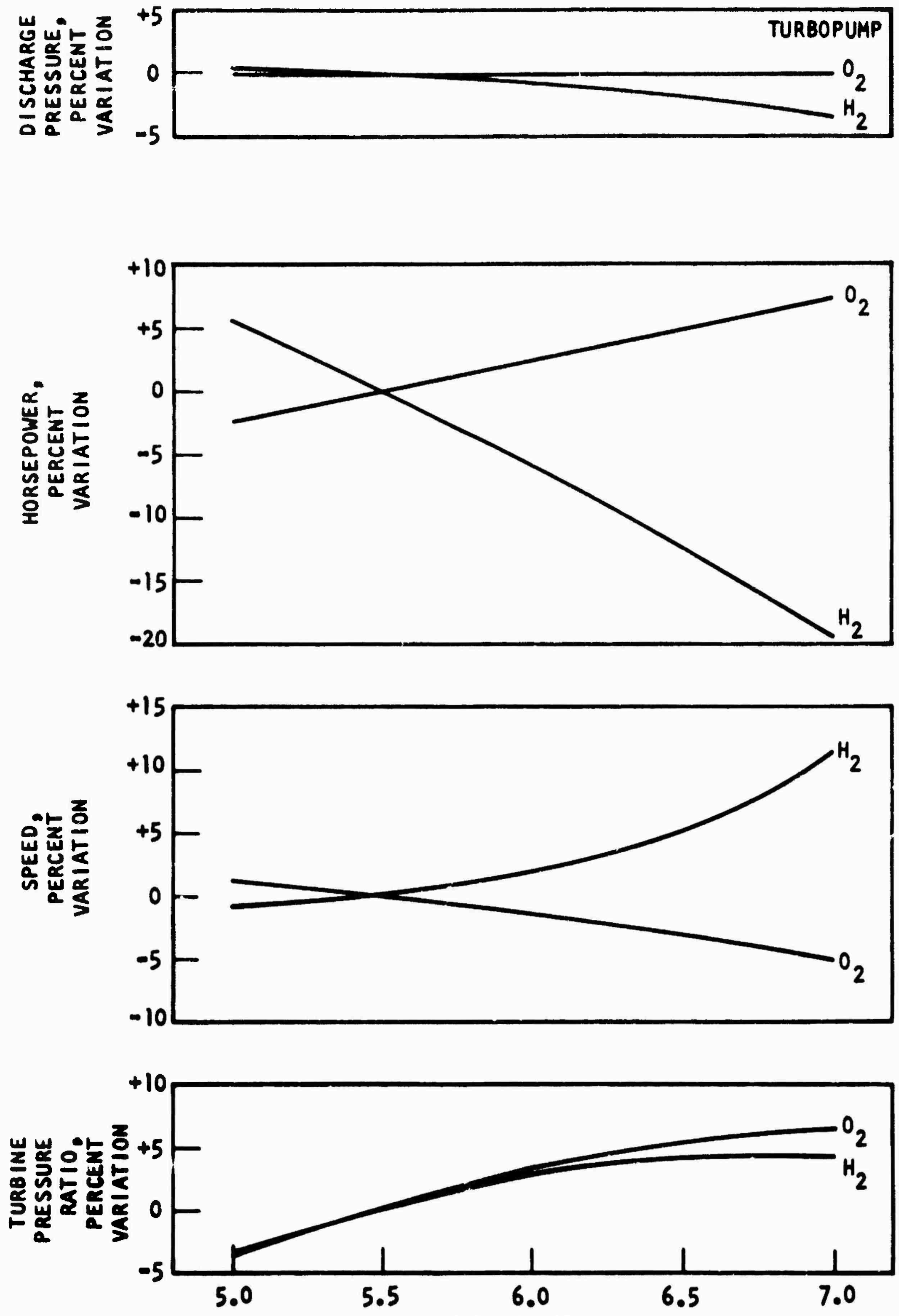


Figure 127. Effect of Design Engine Mixture Ratio on Double-Panel Turbomachinery Parameters

Demonstrator Engine Program Schedule and Costs

The demonstrator engine program cost decreases with increasing mixture ratio as indicated in Fig. 128. Three factors contribute to this trend. First, the decreasing area ratio reduces the thrust chamber assembly fabrication and engineering cost. Second, the turbomachinery fabrication and engineering costs decrease. The oxidizer pump increases in size with increasing mixture ratio, but the larger hydrogen pump decreases in size, resulting in an overall reduced turbomachinery cost. The final effect is the propellant cost. Increasing mixture ratio decreases the requirement for the relatively expensive hydrogen while increasing the amount of the cheaper oxygen. The total propellant cost, therefore, decreases as mixture ratio is increased.

The component differences resulting from the mixture ratio variation will not significantly affect the design or fabrication times required. The types of tests required and test frequencies will not be affected by mixture ratio. Consequently, no change in the demonstrator engine program schedule occurs with the mixture ratio variation.

Development Program Schedule and Costs

The development program cost will decrease with increasing mixture ratio as indicated in Fig. 128. As in the demonstrator engine program, the decreasing thrust chamber size, the reduced turbomachinery size, and the lower propellant cost also are responsible for this trend. Also, as in the demonstrator engine program, no change in the program schedule is required.

First Production Unit Cost

Figure 128 includes the first unit cost trend with mixture ratio. The decrease in thrust chamber area ratio and the reduced total turbomachinery weight with increasing mixture ratio result in a reduction of the cost of fabricating the engine.

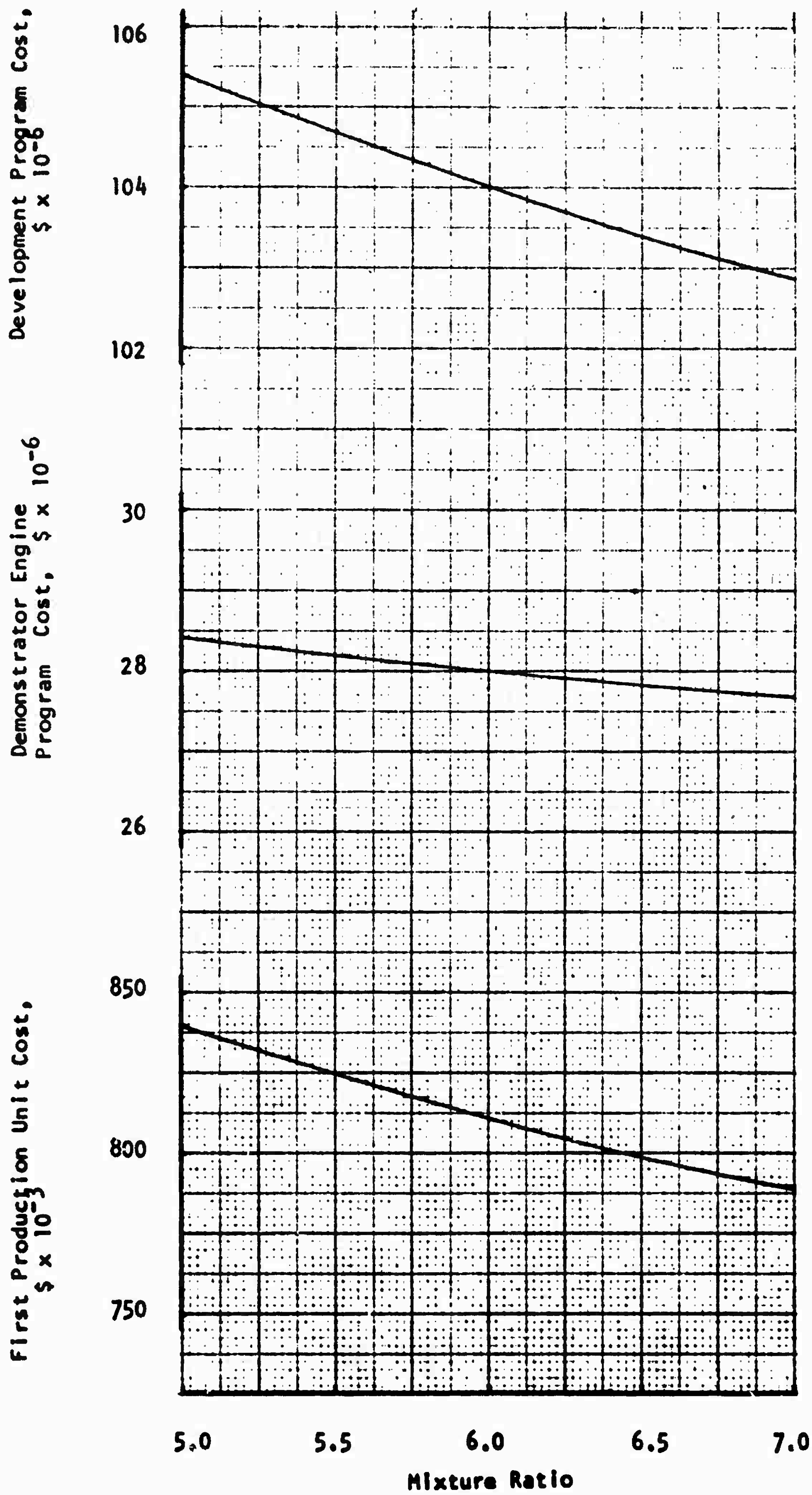


Figure 128. Effect of Design Engine Mixture Ratio on Double-Panel Development Schedule and Costs

VARIATION OF DESIGN FUEL PUMP NPSH

The baseline fuel pump NPSH is 60 feet. The effects of changing this design specification to zero feet or in the range from 15 to 60 feet were determined and the results are presented.

A reduction in design NPSH to approximately 50 feet can be accomplished with relatively minor changes in turbopump design parameters. At a value of 50 feet, the power limit for 12 percent turbine bypass flow is reached. A further reduction in design NPSH to approximately 45 feet can be attained by changing turbopump design parameters and reducing the bypass flow to 7.5 percent of the total fuel flow. A design NPSH below 45 feet requires an additional low-pressure pump. Electric motor and hydrogen gas turbine drives were considered for the low-pressure pump.

Weight

The effect of varying the design fuel pump NPSH on engine weight is shown in Table 58 to be minimal. As design NPSH is reduced from 60 to 50 feet, the engine weight increases slightly. The weight increase is predominantly due to an increase in the size of the fuel pump, which results from a reduction in speed and an increase in horsepower.

TABLE 58. EFFECT OF FUEL PUMP NPSH ON DOUBLE PANEL ENGINE WEIGHT

NPSH	60	50	45	40 to 15, 0
Design Changes		Turbopump Design	Turbopump Design Bypass Flow (7.5%)	Turbopump Design Low-pressure Pump
Weight	398	407	408	407

Engine weight does not vary significantly as NPSH is further reduced from 50 to 45 feet. Although the pump speed decreases, the effect on pump diameter is

compensated for by a reduction in horsepower. The reduced fuel pump horsepower results from a lowering of pump discharge pressure for the higher turbine flows and lower turbine pressure ratios.

A low-pressure pump is required for a design NPSH of less than 45 feet. The weight of the low-pressure pump and electric motor or hydrogen gas turbine drive is offset by the reduction in weight of the main fuel pump, which operates at a relatively high speed and low horsepower because of its high inlet NPSH. The weight increase from the nominal value is due to the larger propellant inlet valves.

Interface Dimensions

As the fuel pump NPSH is reduced from 60 to 45 feet, the main fuel valve inlet diameter increases from 2.5 to 2.6 inches. At lower values of NPSH, the fuel inlet diameter is 3.1 inches. The oxidizer inlet duct is unaffected.

Engine System Operating Conditions

Turbine drive cycle and turbopump assembly operating conditions are not affected substantially as the fuel pump design NPSH is reduced to 50 feet. A reduction in design pump speed is required, which lowers the efficiency and increases the fuel pump horsepower requirement. The increased horsepower requirement is met by altering the flow split to the parallel turbines and increasing the turbine pressure ratios. A comparison of design operating conditions as a function of fuel pump NPSH is presented in Table 59.

Further reductions in design NPSH can be achieved by increasing the available turbine power or reducing pump power requirements. Because the turbine inlet temperature is limited by cooling requirements, the only practical method of increasing the available turbine power is to increase the turbine flow. Increasing the turbine flow (reducing turbine bypass flow to 7.5 percent), in conjunction with changes in turbopump design parameters, is capable of meeting the power requirements for an NPSH of approximately 45 feet.

TABLE 59. EFFECT OF DOUBLE-PANEL FUEL PUMP DESIGN NPSH ON TURBOPUMP OPERATING CONDITIONS

Fuel Pump NPSH, feet	60		50		45		40 to 15, 0	
Turbine Bypass Flow percent	12		12		7.5		12	
Propellant	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2
Low-pressure Pump	No	No	No	No	No	No	No	
Main Turbopump Parameters*								
Discharge Pressure, psia	0.0	0.0	0.0	+5.7	0.0	+0.6	0.0	-7.2
Horsepower, hp	0.0	0.0	-0.9	+16.5	-0.3	+13.5	-0.9	-23.2
Speed, rpm	0.0	0.0	-5.1	-10.6	-0.6	-15.7	-6.1	+36.9
Turbine Pressure Ratio	0.0		+11.8	+11.8	+2.6	+ 1.2	-9.2	-11.5

*Values shown are percent variations from design values

For a design NPSH of less than 45 feet, pump power requirements must be reduced by using a low-speed, low-pressure fuel pump to allow the main fuel pump to operate at a higher efficiency. Although the optimum main fuel turbopump design requires an NPSH of less than 250 feet, a head of approximately 370 feet must be developed by the low-pressure pump to maintain low-pressure pump parameters within design limits. Because of these design limits, low-pressure pump parameters do not vary significantly as a function of inlet NPSH.

For a low-pressure pump head of approximately 370 feet, the horsepower requirement is so small as to make a gas-turbine drive impractical. Alternatives are the use of an electric motor drive or an increase in low-pressure pump head to 1250 feet and use of a hydrogen-driven gas turbine in parallel with the main turbines. Low-pressure pump parameters for an inlet NPSH of zero are shown in Table 60 for both types of drives. The low-pressure pump is designed for 25-percent vapor pumping capability. The choice between the low-pressure pumps does not significantly change the main fuel pump head requirement and, therefore, only one design for the main pump was presented in Table 60.

TABLE 60. DOUBLE-PANEL FUEL LOW-PRESSURE PUMP DESIGN PARAMETERS

Low-Pressure Pump Drive	Electric Motor	H_2 Gas Turbine
Head, feet	371	1250
Power, hp	8.6	32
Speed, rpm	25,800	38,800

Start Transient

Variations in the start time to 90 percent thrust as a function of fuel pump NPSH are expected to be small. The largest variation anticipated would occur with the use of a low-pressure pump if "time constants" associated with rotating inertias of the low-pressure pump and main fuel turbopump are significantly different from that of the baseline fuel turbopump. The variation would occur during the initial acceleration of the fuel turbopump under open-loop control. The maximum effect is expected to be ±0.3 second for a baseline engine start time of 4.4 seconds.

Complexity

Engine system complexity is not affected until the design NPSH is reduced below 45 feet and a low-pressure pump is required. Addition of a low-pressure pump and either an electric motor drive or a turbine with associated gas flow duct in parallel with the main turbines increases system complexity. A gas turbine drive also may require an open-loop control valve to prevent overspeed during start if the low-pressure turbopump response is significantly faster than the main fuel turbopump.

Technology Improvements Needed

No technology improvements are needed for the hydrogen pump inlet conditions considered in this study. Satisfactory low pressure and main pump designs can be achieved to meet inlet NPSH's of 16 to 60 feet with present inducer technology. For a zero-

NPSH specification, the low-pressure pump must be designed with two-phase pumping capability. Two-phase pumping has been investigated under two recent NASA contracts (Ref. 5 and 6). Analytical techniques have been developed to design and predict vapor pumping capability, and good agreement exists between these analyses and experimental results using hydrogen pumps with up to approximately 35 percent vapor, by volume, at the inlet. If higher capability were needed, the state of the art could be advanced by investigating higher overall incidence-to-blade angle ratios.

Demonstrator Engine Program Schedule and Costs

As hydrogen pump NPSH is decreased from the nominal 60 to 45 feet, the required increase in the size of the hydrogen pump results in an increase in the turbomachinery-related engineering and the turbomachinery fabrication costs. For an NPSH in the range of 40 to 15 feet, the turbomachinery is slightly lighter than nominal, but the increased complication of adding a hydrogen low-pressure pump results in additional engineering and fabrication costs as well as increased pump test costs. The inlet diameter of the hydrogen low-pressure pump establishes a need for larger flow area main inlet valves which also add to the cost of this system.

The hydrogen low-pressure pump design for an NPSH between 40 and 15 feet also is adequate for operation at an NPSH of zero feet. To obtain zero NPSH capability, however, the turbomachinery and engine system test program costs will increase because additional testing to evaluate and demonstrate vapor pumping capacity is required. For the testing, the two-phase propellant would be generated by a Frantz screen installed immediately upstream of the pump inlet. The vapor fractions can be determined from measurements of pressure and temperature in the pure liquid upstream of the screen and in the two-phase fluid downstream of the screen. This technique has been successfully used on previous programs without testing problems and has a relatively minor impact on facility costs. The additional engine system testing increases the demonstrator engine program duration by 0.25 month when a pump inlet NPSH of zero feet is required. The cost changes in the demonstrator engine program because of lower fuel pump NPSH values are presented in Table 61.

TABLE 61. EFFECT OF DOUBLE-PANEL FUEL PUMP NPSH ON DEVELOPMENT SCHEDULE AND COSTS

	NPSH, ft			
	60 (Nom.)	45	40 → 15	0
Demonstrator Program Cost, $	28.2 M	28.4 M	28.7 M	29.1 M
Δ Cost, $	0	+ .2 M	+ .5 M	+ .9 M
Demonstrator Program Length, Months	31	31	31	31.25
Δ Length, Months	0	0	0	+ .25
Development Program Cost, $	104.7 M	105.2 M	106.3 M	107.9 M
Δ Cost, $	0	+ .5 M	+1.6 M	+3.2 M
Development Program Length, Months	55	55	55	56.4
Δ Length, Months	0	0	0	+1.4
First Unit Cost, $	825 K	835 K	846 K	846 K
Δ Cost, $	0	+10 K	+21 K	+21 K

Development Program Schedule and Costs

The cost increases in the development program which result from reduced hydrogen pump NPSH values also are presented in Table 61. The same factors which contributed to the cost increase in the demonstrator program also are responsible for the development program cost increase.

First Production Unit Cost

The increased fabrication cost resulting from the engine design changes necessary to meet lower hydrogen pump NPSH values are summarized in Table 61.

VARIATION OF DESIGN OXIDIZER PUMP NPSH

The baseline oxidizer pump design NPSH is 16 feet. The effects of varying this design specification to zero feet and over a range from 2 to 16 feet were determined and the results are presented. The general approach to this study was essentially the same as for the hydrogen pump NPSH variation. A reduction in design oxidizer pump NPSH to approximately 12 feet can be accomplished with relatively minor changes in turbopump design parameters. At a value of 12 feet, the power limit for 12 percent turbine bypass flow is reached. A further reduction in design NPSH to approximately 8 feet can be attained by changing turbopump design parameters and reducing the bypass flow to 7.5 percent of the total fuel flow. A design NPSH below 8 feet requires a low-speed oxidizer low-pressure pump. Electric motor and hydrogen gas turbine drives were considered for the low-pressure pump.

Weight

The effect of varying the design oxidizer pump NPSH on engine weight is shown in Table 62. As design NPSH is reduced from 16 to 8 feet, the weight increases slightly, primarily due to an increase in the sizes of the oxidizer pump and the propellant inlet valves (common valves used). The increase in the size of the pump results from a reduction in speed and an increase in horsepower.

An oxidizer low-pressure pump is required for a design NPSH of less than 8 feet, but the increase in engine weight is primarily due to the larger inlet valves. A portion of the weight of the low-pressure pump and electric motor or hydrogen gas turbine drive is offset by the reduction in weight of the main oxidizer pump which operates at a relatively high speed and low horsepower because of its high inlet NPSH.

TABLE 62. EFFECT OF DOUBLE-PANEL OXIDIZER PUMP NPSH ON ENGINE WEIGHT

NPSH	16	12	8	6→2, 0
Engine System Changes	-	Turbopump Design	Turbopump Design	Turbopump Design
			Bypass (7.5)	Low-pressure pump
Weight	398	407	423	434

Interface Dimensions

As the oxidizer pump NPSH is reduced from 16 to 11 feet, the inlet diameter of the main oxidizer valve is unchanged (3.0 inches). As NPSH is further reduced to 8 feet, the diameter increases to 3.2 inches. At lower values of NPSH, the oxidizer inlet diameter is 4.4 inches. The fuel valve inlet diameter is unaffected (2.5 inches).

Engine System Operating Conditions

As oxidizer pump design NPSH is reduced to 12 feet, a reduction in oxidizer pump speed is required, which lowers the efficiency and increases the horsepower slightly. More important is the altering of the flow split to the parallel turbines which increases the fuel turbine pressure ratio and fuel pump head requirement. A comparison of design operating conditions for the baseline engine (oxidizer NPSH = 16 feet) and those for the range of NPSH values from zero the 16 feet is presented in Table 63.

TABLE 63. EFFECT OF DOUBLE-PANEL OXIDIZER PUMP DESIGN NPSH ON TURBOPUMP OPERATING CONDITIONS

Oxidizer Pump NPSH, feet	16		12		8		6 to 2,0	
Turbine Bypass Flow, percent	12		12		7.5		12	
Propellant	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2
Low-Pressure Pump	No	No	No	No	No	No	Yes	No
Main Turbopump Parameters*								
Discharge Pressure, psia	0.0	0.0	0.0	+2.5	0.0	+5.0	0.0	-4.1
Horsepower, hp	0.0	0.0	+0.9	+1.3	+9.2	+5.1	-22.2	-7.6
Speed, rpm	0.0	0.0	-19.2	+4.3	-40.6	+3.3	+47.3	+4.5
Turbine Pressure Ratio	0.0	0.0	+1.3	+4.2	+11.1	+7.9	-5.2	-6.1

*Values shown are percent variation from design value

Further reductions in design NPSH can be achieved by increasing the available turbine power or reducing pump power requirements. Because the turbine inlet temperature is limited by cooling requirements, the only practical method of increasing the available turbine power is to increase the turbine flow. Increasing the turbine flow (reducing turbine bypass flow to 7.5 percent), in conjunction with changes in turbopump design parameters, is capable of meeting the power requirements for an NPSH of approximately 8 feet.

For a design NPSH of less than 8 feet, pump power requirements must be reduced by using a low-speed oxidizer low-pressure pump to allow the main oxidizer pump to operate at a higher efficiency. A main pump NPSH of approximately 70 feet is necessary to achieve an optimum main pump design; therefore, a low-pressure pump head of 70 feet is required for an inlet NPSH of zero, and a correspondingly lower head must be developed for design inlet NPSH's up to 8 feet. Low-pressure pump design parameters over this range of inlet NPSH's do not vary significantly, and only the zero NPSH design is presented in Table 64. The low-pressure pump is designed for 25 percent vapor pumping capability.

TABLE 64. DOUBLE-PANEL OXIDIZER LOW-PRESSURE PUMP DESIGN PARAMETERS

Low-Pressure Pump Drive	Electric Motor	H_2 Gas Turbine
Head, feet	70	125
Power, hp	8.7	18
Speed, rpm	4470	5960

For an oxidizer low-pressure pump with a head of 70 feet or less, the horsepower requirement is so small as to make a gas-turbine drive impractical. Alternatives are the use of an electric motor drive or an increase in head to 125 feet and use of a hydrogen-driven gas turbine in parallel with the main turbines. The choice between low-pressure pumps does not significantly change the main oxidizer pump design since both provide at least the NPSH required for an optimum main pump design (70 feet).

Start Transient

The start transient is not significantly affected by variations in oxidizer pump NPSH for the range of values considered. The oxidizer turbopump acceleration is limited by the desired rate of increase in chamber pressure rather than the influence of NPSH on size and, therefore, rotating inertia of the pump.

Complexity

Engine system complexity is not affected until the design NPSH is reduced below 8 feet and an oxidizer low-pressure pump is required. Addition of a low-pressure pump and either an electric motor drive or a turbine with associated gas flow duct in parallel with the main turbines increases system complexity. A gas turbine drive also would require an open-loop control valve at the inlet of the oxidizer low-pressure pump turbine to prevent flow to this turbine during initial acceleration of the fuel turbopump.

Technology Improvements Needed

Technology is not now available for pumping two-phase oxygen, but a program is in progress to establish design criteria and test two-phase oxygen inducers (Contract NAS8-26645). Analytical techniques and test experience with two-phase hydrogen will be utilized in the program.

Demonstrator Engine Program Schedule and Costs

As oxygen pump NPSH is decreased from the nominal 16 to 8 feet, the required increase in the size of the oxygen pump results in an increase in the turbomachinery-related engineering and the turbomachinery fabrication costs. For the range of NPSH between 6 and 2 feet, there also is the increased complication of adding an oxygen low-pressure pump which results in additional engineering and fabrication costs as well as increased pump test costs. The inlet diameter of the oxygen pump or low-pressure pump increases with decreasing NPSH which establishes a need for larger flow area main inlet valves which also add to the cost of the lower oxygen pump NPSH systems.

The oxygen low-pressure pump design for an NPSH between 6 and 2 feet also is adequate for operation at zero feet NPSH. To obtain zero-NPSH capability, however, the turbomachinery and engine system test program costs will increase because additional testing to evaluate and demonstrate vapor pumping capacity are required.

The cost changes in the demonstrator engine program because of lower oxidizer pump NPSH values are presented in Table 65.

Development Program Schedule and Costs

The cost increases in the development program which result from reduced oxygen pump NPSH values also are presented in Table 65. The same factors which contributed to the cost increase in the demonstrator program also are responsible for the development program cost increases.

TABLE 65. EFFECT OF DOUBLE-PANEL OXIDIZER PUMP NPSH ON DEVELOPMENT SCHEDULE AND COSTS

	NPSH, feet				
	16 (Nom.)	12	8	6 → 2	0
Demonstrator Program Cost, $	28.2 M	28.3 M	28.6 M	29.1 M	29.5 M
ΔCost, $	0	+ .1 M	+ .4 M	+ .9 M	+1.3 M
Demonstrator Program Length, Months	31	31	31	31	31.25
ΔLength, Months	0	0	0	0	+ .25
Development Program Cost, $	104.7 M	105.1 M	106.2 M	107.6 M	109.2 M
ΔCost, $	0	+ .4 M	+1.5 M	+2.9 M	+4.5 M
Development Program Length, Months	55	55	55	55	56.4
ΔLength, Months	0	0	0	0	+1.4
First Unit Cost, $	825 K	833 K	854 K	873 K	873 K
ΔCost, $	0	+ 8 K	+29 K	+48 K	+48 K

First Production Unit Cost

The fabrication cost increases resulting from the engine design changes necessary to meet lower oxygen pump NPSH values are summarized in Table 65 .

THERMAL CYCLES (REUSABLE VERSION)

The baseline engine system (reusable version) is designed for 300 thermal cycles between overhauls. The effects of changing this design specification to 60 and 600 were investigated.

Nozzle Area Ratio

A reduction in the thermal cycle design specification to 60 thermal cycles does not have any impact on the engine system because any restrictions on operating conditions which might be relaxed also are constrained by the 10-hour life specification. Increasing the thermal cycle specification to 600, however, requires a reduction in the thrust chamber wall temperatures. The optimum mission performance criteria for these more restrictive heat transfer requirements results in a reduction in the nozzle area ratio, but the chamber pressure (1000 psia) is not affected. The nozzle area ratio for the higher cycle life is 126, which is considerably lower than the baseline value of 200.

Specific Impulse

The effect of varying the thermal cycle requirement on delivered specific impulse is substantial because of the variation in area ratio. As the specification is increased from 300 to 600 cycles, specific impulse decreases from 471. 1 to 462.8 seconds.

Engine Weight

The effect of cycle life on engine system dry weight is primarily due to the variation in area ratio, which directly influences the engine diameter. Engine weight decreases from 398 to 349 pounds as the life is increased from 300 to 600 cycles.

Engine Cooling Requirements

The higher cycle life capability is accomplished by reducing the temperature gradient from the hot-gas wall to the coolant channel closure wall. A reduction in the hot-gas wall temperature of approximately 245 R is required.

Engine Maintenance Requirements

Engine maintenance requirements are not affected by variations in the number of thermal cycles over the range considered.

Engine System Operating Conditions

Engine system operating conditions are unchanged if the thermal cycle requirement is reduced. Operating conditions for a 600-cycle specification are summarized in Table 66 . The turbine bypass flow was maintained at 12 percent of the total fuel flow.

TABLE 66. EFFECT OF THERMAL CYCLES ON DOUBLE-PANEL TURBOPUMP OPERATING CONDITIONS

Thermal Cycles	300		600	
Propellant	O_2	H_2	O_2	H_2
Main Turbopump Parameters*				
Discharge Pressure, psia	0.0	0.0	0.0	+2.3
Horsepower, hp	0.0	0.0	+2.1	+2.3
Speed, rpm	0.0	0.0	+1.8	+4.1
Turbine Pressure Ratio	0.0	0.0	+16.2	+14.4

*Values shown are percent variations from design values

Technology Improvements Needed

No technology improvements are needed to provide a lower thermal cycle capability. However, to increase the engine thermal cycle capability, operating hot-gas wall temperatures must be reduced. The effect of the reduced wall temperature on other engine system and component operating conditions can be minimized if more detailed and exact thermal cycle fatigue and creep data were available for the specific materials that are to be used in the thrust chamber fabrication.

Demonstrator Engine Program
Schedule and Costs

Thermal cycles will be accumulated by on/off cycling as part of specific tests. The number of cycles per test can be adjusted to account for the variations in thermal cycle design requirements without altering the number of tests required or the test costs. The difference in program cost is consequently affected only by the engineering and fabrication cost variations resulting from changes in the actual engine design. Because the engine design is unchanged by the reduction in thermal cycles to 60, the cost is unchanged. The reduction in area ratio necessary for the 600 thermal cycles reduces the demonstrator engine program costs by 0.4 M dollars from the nominal 28.2 M dollars to 27.8 M dollars. The program schedule is unchanged by the thermal cycle variations.

Development Program Schedule and Costs

As in the demonstrator engine program,the only cost change results from the area ratio decrease at 600 thermal cycles. The cost decreases by 1.1 M dollars from 104.7 M dollars to 103.6 M dollars. The program schedule is unchanged by the thermal cycle variations.

First Production Unit Cost

The 126 area ratio engine first production unit cost is 795 K dollars, lower by 30 K dollars than the nominal 200 area ratio engine cost of 825 K dollars.

THERMAL CYCLES (EXPENDABLE)

An increase in the number of required thermal cycles from 6 to 10 for the ground-based expendable version would not affect the engine system design, performance, or cost because it is designed for considerably more than 10 cycles for use in the development program.

NUMBER OF VACUUM STARTS

The nominal engine system is capable of 60 vacuum starts without requiring inspection or component servicing. A variation of this specification to 20 and 600 was investigated.

The engine system is designed to a maximum thrust chamber wall temperature which is consistent with the 10-hour life and the 300 thermal cycle specifications. Because lowering the vacuum start requirement does not affect either of these specifications, there is no resulting relaxation of restrictions on operating conditions. Therefore, a reduction of the requirement to 20 vacuum starts does not affect engine system or component design, performance, or cost.

Increasing the vacuum start capability to 600 necessarily implies an increase in the cycle life specification to 600 thermal cycles. The impact on the engine system is, therefore, the same as described in the section on Thermal Cycles (reusable version).

HOUR LIFE

The baseline engine system has a life specification of 10 hours between overhauls. Engine design is rather insensitive to variations in this requirement over a range from 2 to 20 hours, however, and performance and operating conditions are not affected.

Demonstrator Engine Program Schedule and Costs

Though the engine design is not affected by the life requirement, the life demonstration oriented testing is and, consequently, the costs for test operations, test consumables, and test support engineering change with engine life. This effect on the demonstrator engine program cost is presented in Fig. 129.

Development Program Schedule and Costs

The variation in test requirements with engine life for the development program are summarized in Fig. 130. These requirements establish the effect on development program length which is presented in Fig. 131. The cost as a function of life for the development program is included with the demonstration program cost in Fig. 129.

First Production Unit Cost

Because the engine design is insensitive to change in the life requirement, the cost of fabricating the first production unit is constant with engine life.

MAXIMUM RUN TIME

The maximum run time specification is 1000 seconds. The effects on engine design and costs of changing this requirement to 500 to 2000 seconds are insignificant for fixed thermal cycle and hour-life requirements.

MAXIMUM ORBIT STORAGE TIME

The baseline engine system is designed for a maximum orbit storage time of 52 weeks. The primary consideration is protection against the possibility of vacuum cold welding, especially of the turbopump bearings. This protection will be provided by a regular rotation of the turbopumps or by pressurization of the

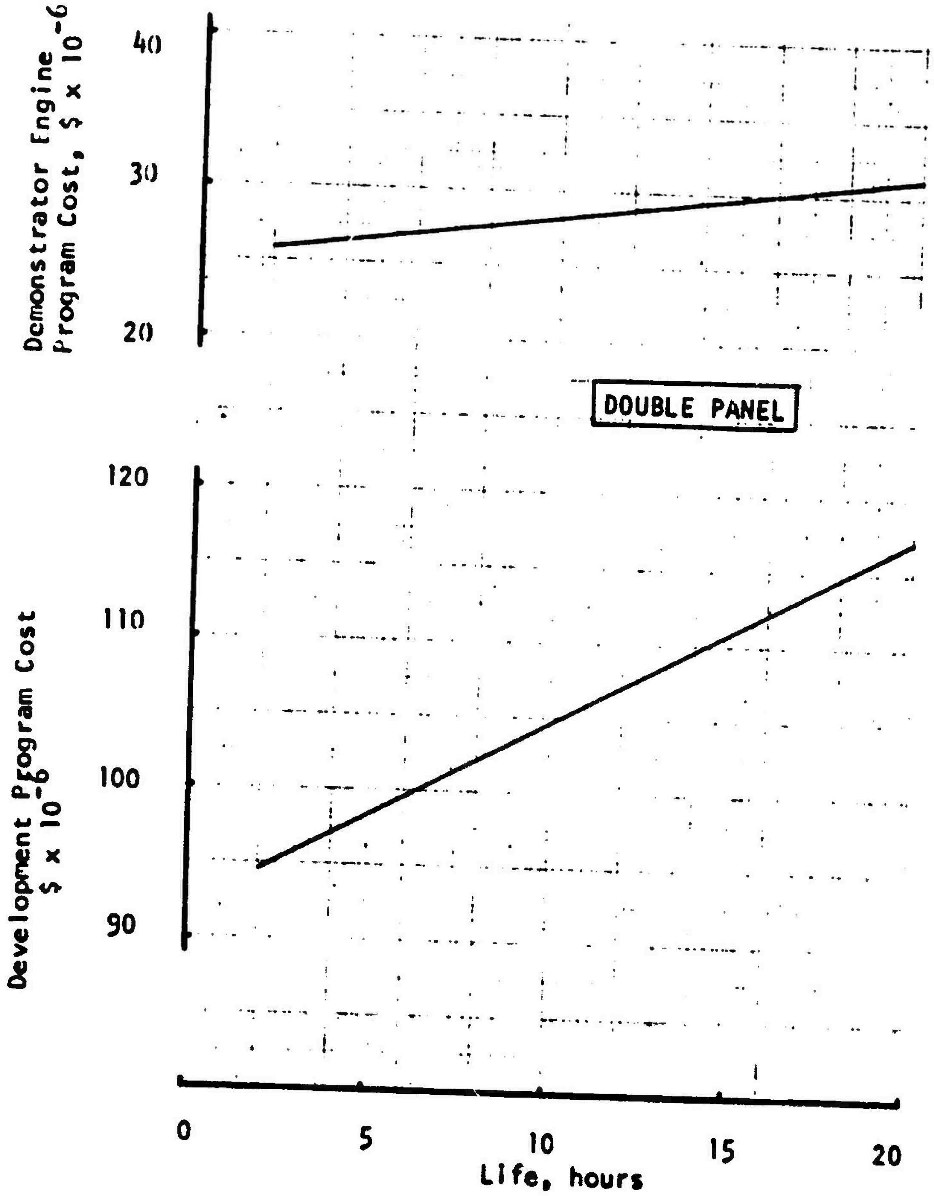


Figure 129. Effect of Life on Cost

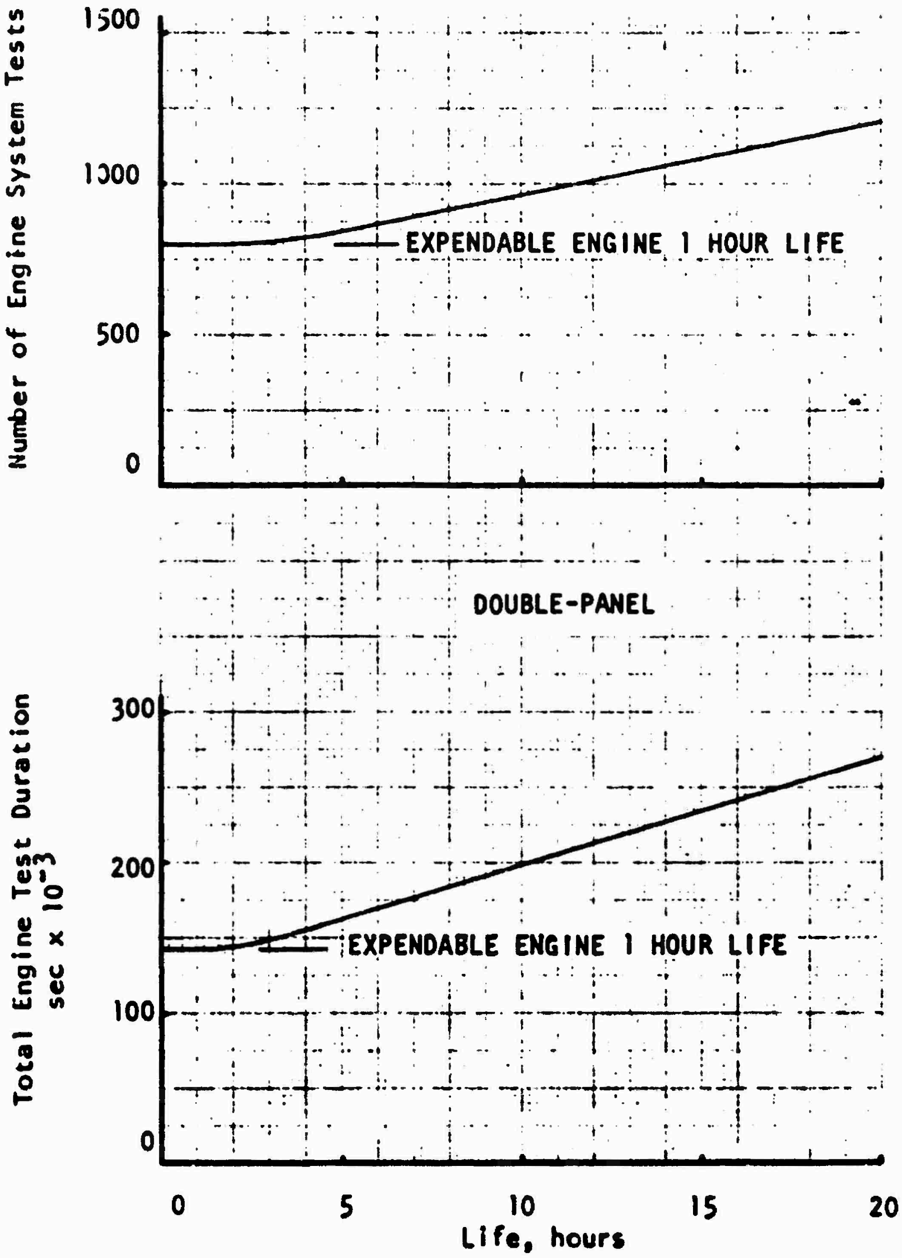


Figure 130. Effect of Life on Development Program Engine Testing

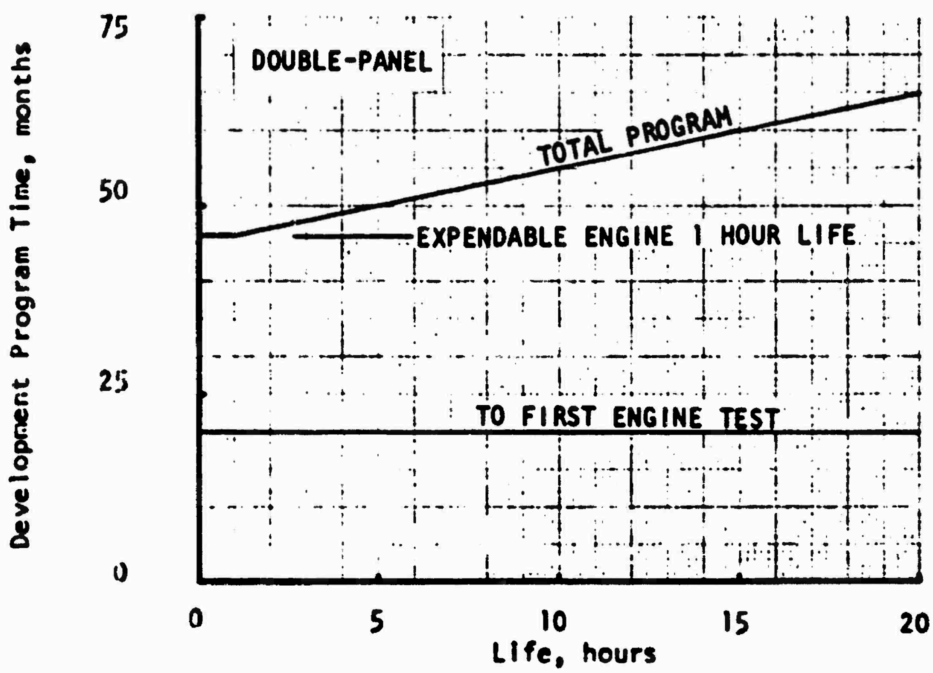


Figure 131. Effect of Life on Development Program Duration

engine feed system. Both techniques would utilize an inert gas from an external source. Pressurization of the feed system would require sealing of the thrust chamber throat. Based on the present knowledge of the vacuum cold-welding process, a variation in the maximum storage time specification over a range from 2 to 104 weeks does not significantly affect the engine system design or cost.

GIMBAL ANGLE

The aerospike engine design does not include flexible propellant supply ducting or gimbal actuators. The basic engine is, therefore, unchanged by gimbal angle variations between 3 and 7 degrees. There is no effect on weight, dimensions, gimbal power requirement, or first production unit cost.

GIMBAL ACCELERATION

Weight

Engine system weight will increase by approximately 10 pounds when gimbal acceleration is increased from its nominal value of 5 rad/sec^2 up to 10 rad/sec^2 due to the increased gimbal mount strength required to withstand the increased actuator force and heavier mounts to support the engine components during the higher acceleration.

Interface Dimensions

Engine interface dimensions will not be affected by the variation.

Engine Maintenance Requirements

The engine components are capable of withstanding the increased acceleration without increasing their maintenance requirements.

Gimbal Power Requirements

Gimbal power is increased to 0.91 horsepower, an increase of 0.12 horsepower over the nominal engine requirements.

First Production Unit Cost

The strengthened gimbal and component mounts will have an insignificant effect on the first unit cost.

NOZZLE AREA RATIO

The baseline engine has a nozzle expansion area ratio of 200. The effects of varying this design specification between 100 and 400 were investigated.

Specific Impulse

The effect on vacuum delivered performance of varying nozzle area ratio from 100 to 200 is shown in Fig. 132. The area ratio cannot be increased above 200 because of heat transfer limits.

Weight

The effect of varying nozzle area ratio on engine system weight is shown in Fig. 133. The weight variation is due to the influence of area ratio on engine diameter.

Envelope Dimensions

Envelope dimensions increase as nozzle area ratio increases. Length and diameter data are shown in Fig. 134.

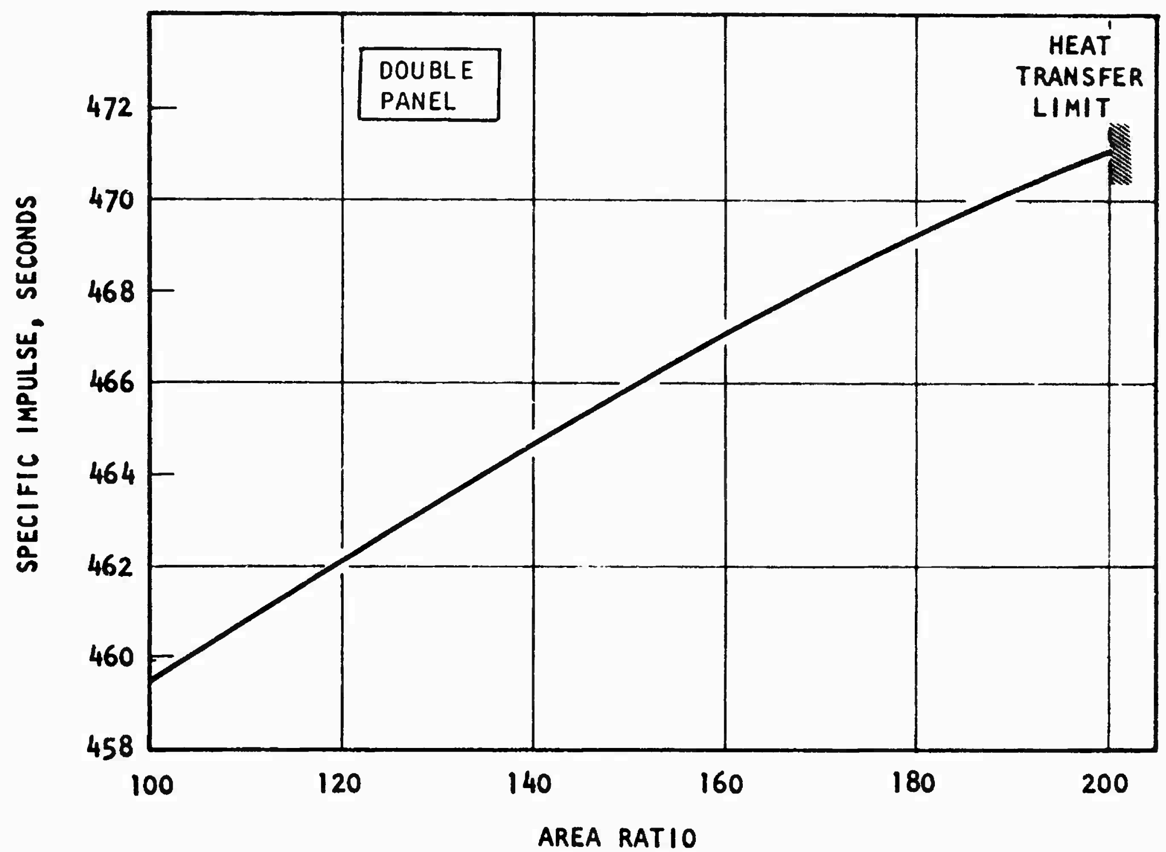


Figure 132. Effect of Nozzle Area Ratio on Delivered Vacuum Engine Performance

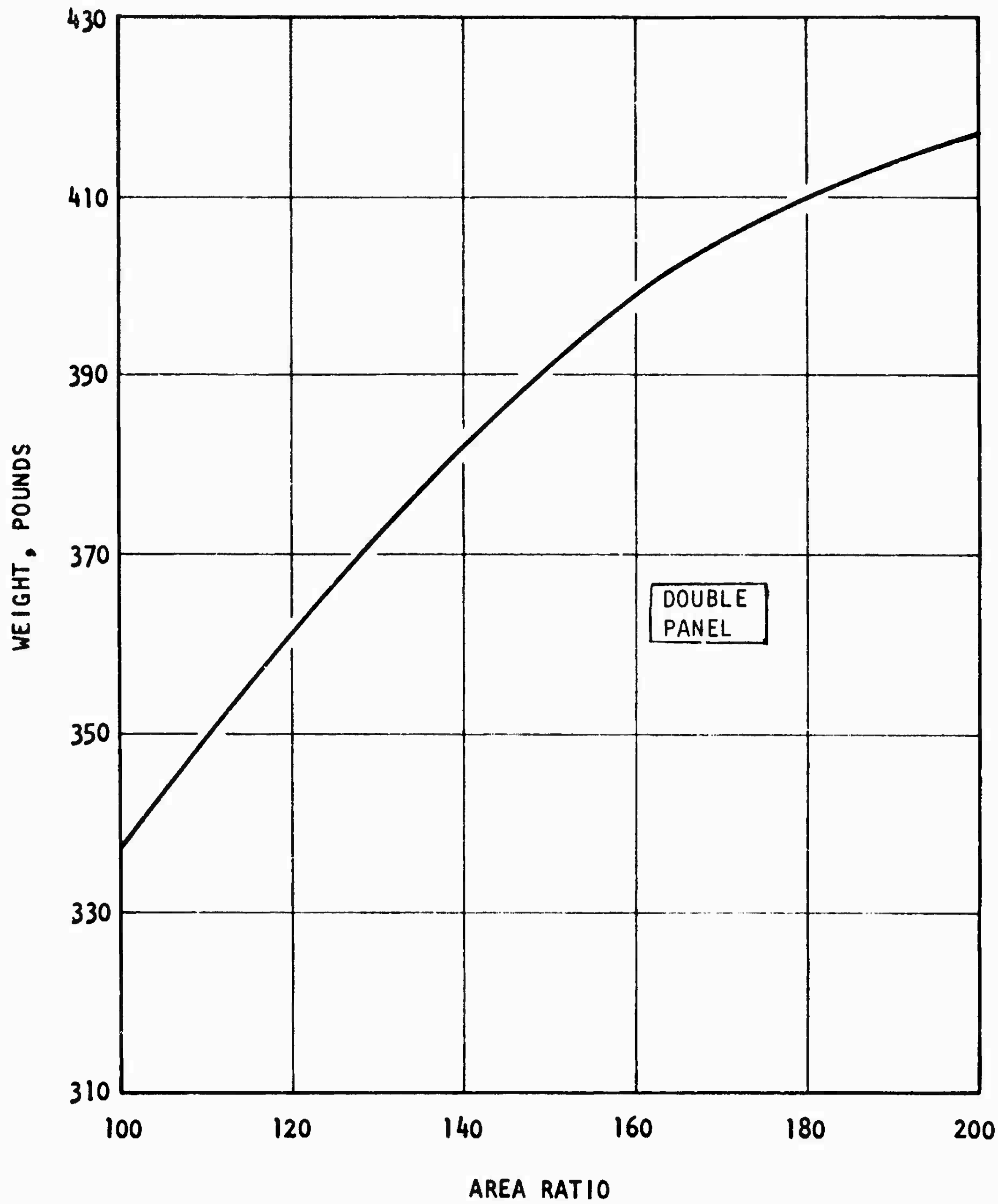
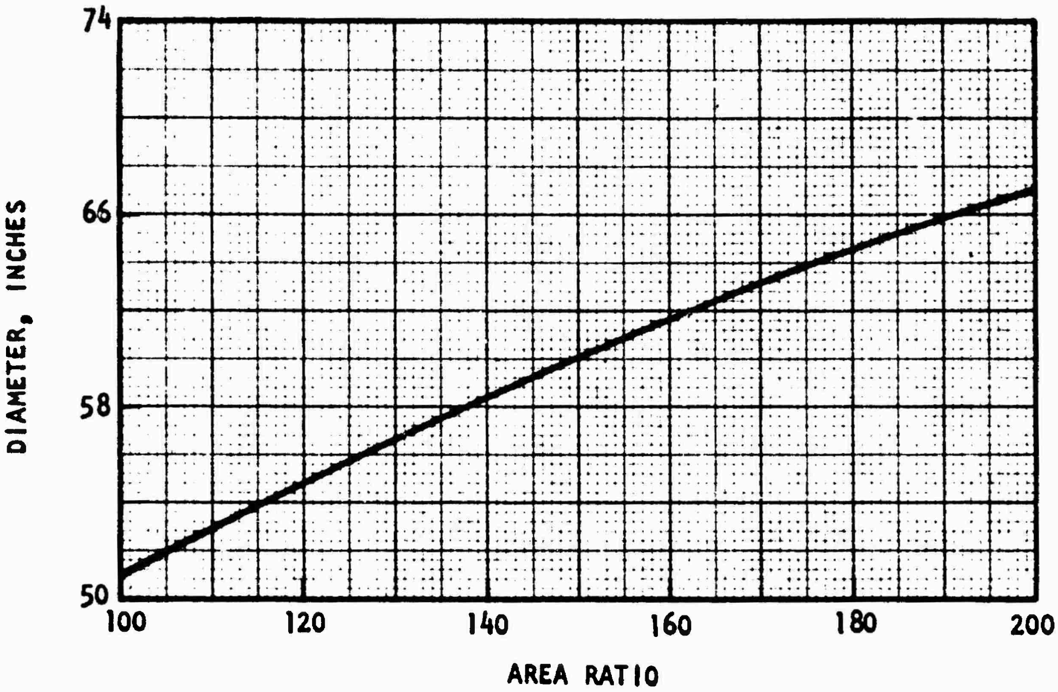


Figure 133. Effect of Nozzle Area Ratio on Engine System Dry Weight



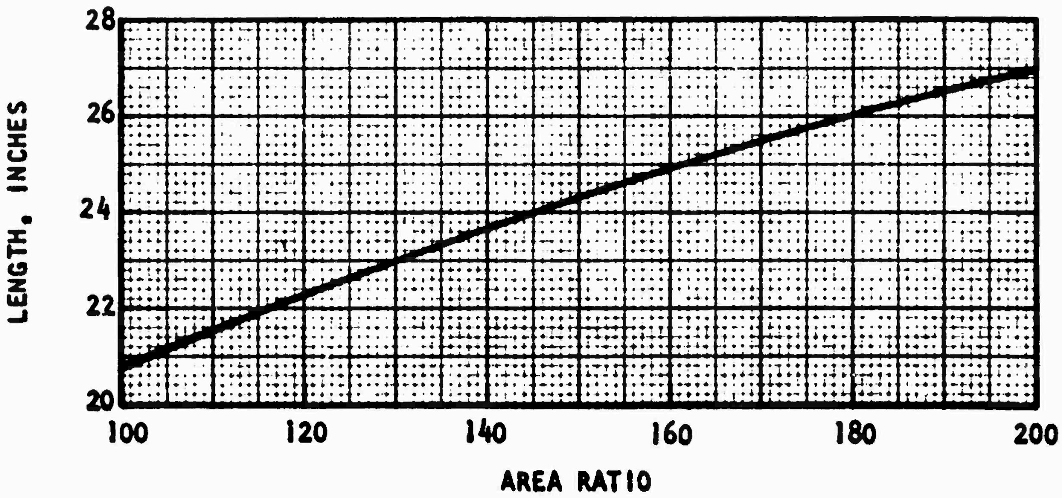


Figure 134. Effect of Nozzle Area Ratio on Double-Panel Engine Envelope Dimensions

Cooling Requirements

The effect of varying nozzle area ratio on jacket exit temperature is shown in Fig. 135.

Gimbal Power Requirements

The gimbal power requirement decreases from its nominal value of about 0.79 horsepower at 200 area ratio to about 0.69 horsepower at 100 area ratio.

Technology Improvements Needed

No technology improvements are needed for nozzle area ratios between 100 and 200. Area ratios greater than 200 exceed heat transfer limits.

Demonstrator Engine Program Schedule and Costs

The demonstrator engine program cost increases with increasing area ratio due to the increased costs of designing and fabricating the thrust chamber assembly. This effect on the cost is illustrated in Fig. 136. The design, fabrication, and test efforts do not change significantly with area ratio so program duration remains constant.

Development Program Schedule and Costs

The effect of area ratio on development cost is also presented in Fig. 136. As in the demonstrator program the duration is unchanged.

First Production Unit Cost

The fabrication cost of the thrust chamber assembly is the only effect on first unit cost of area ratio. This effect is illustrated in Fig. 136.

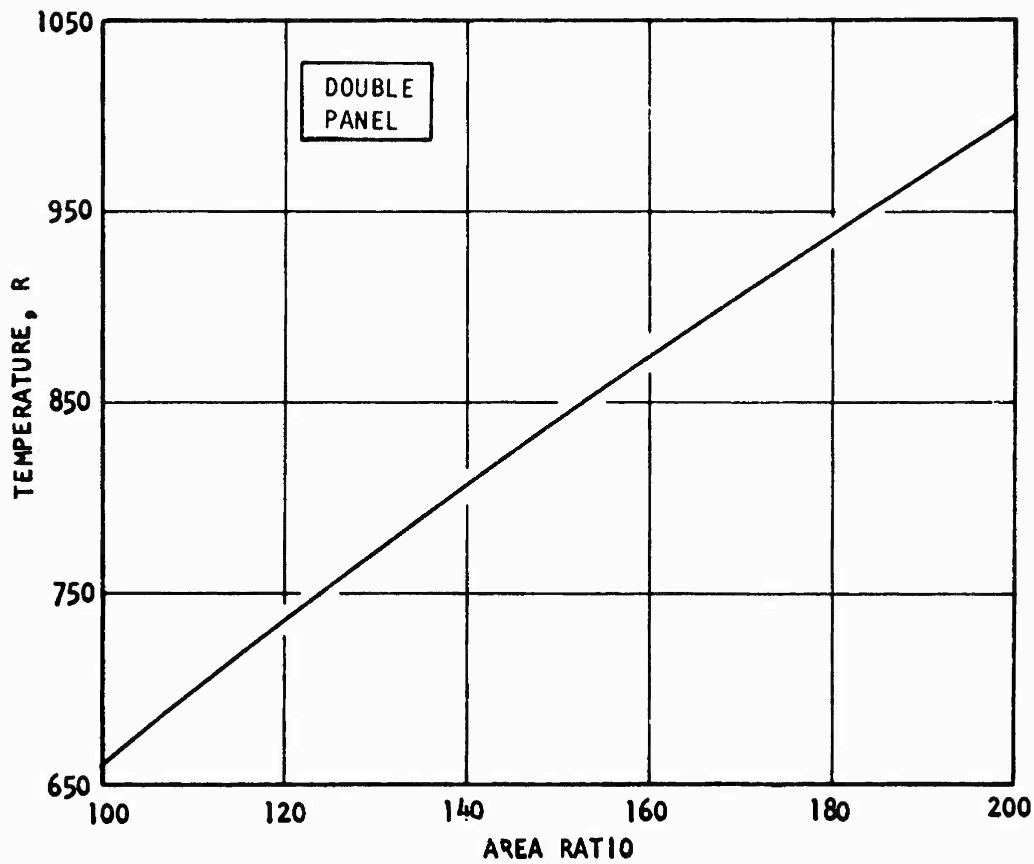


Figure 135. Effect of Nozzle Area Ratio on Jacket Exit Temperature

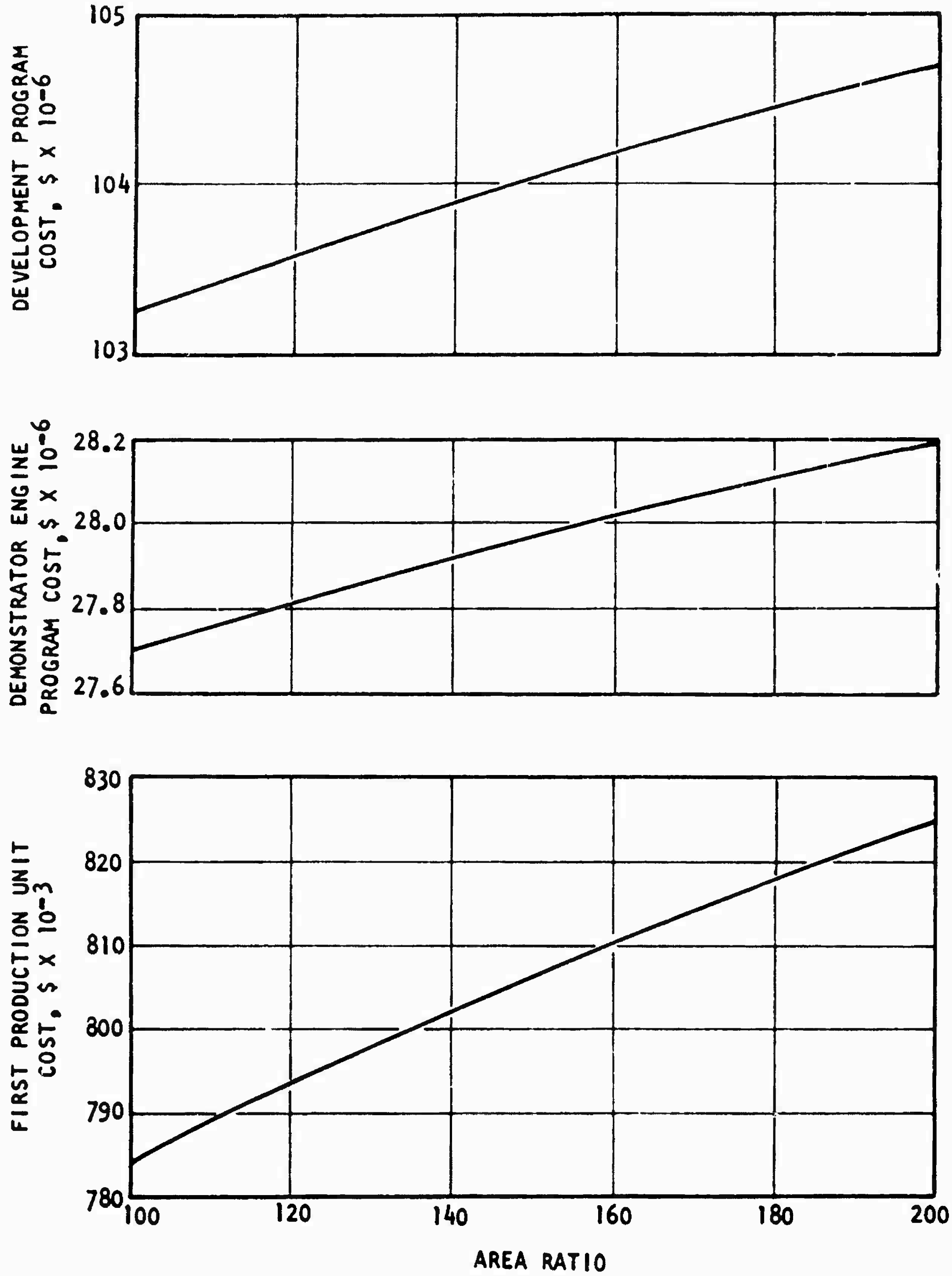


Figure 136. Effect of Nozzle Area Ratio on Double-Panel Costs

VARIATION OF THROTTLE CAPABILITY

The baseline engine has a capability of throttling to 20 percent of design thrust. The effects of eliminating the throttle requirement or extending the capability to 10-percent thrust were investigated.

Specific Impulse

The variation in design engine performance as a function of the required throttle capability is negligible. Thrust chamber heat transfer significantly affects performance of the expander turbine drive cycle, but the double-panel cooling circuit design is not affected by the required throttle range because the most restrictive operating conditions occur at design thrust (25,000 pounds). Therefore, it is not necessary to vary the design nozzle area ratio, chamber pressure, or cooling circuit as the design throttling ratio is varied.

As the engine is throttled, perfromance decreases as shown in Fig. 137. This curve is applicable for all engines to their lower thrust limit.

Weight

Engine system dry weight is affected slightly by throttling capability, primarily due to design changes to the turbopumps.

If the throttling capability is eliminated, the fuel pump discharge pressure can be reduced approximately 25 psi by reducing the oxidizer turbine control valve pressure drop, resulting in an insignificant weight savings. The fuel pump is the only component whose design is influenced by the 5:1 throttling requirement.

Throttling to 10 percent of design thrust would require redesign of the pumps to prevent operation in the positive-slope region of the head-flow performance maps. By designing the pumps for higher-than-nominal head, operation in the positive-slope region of the head-flow maps can be avoided as the engine is throttled. The weight penalty is approximately 11 pounds.

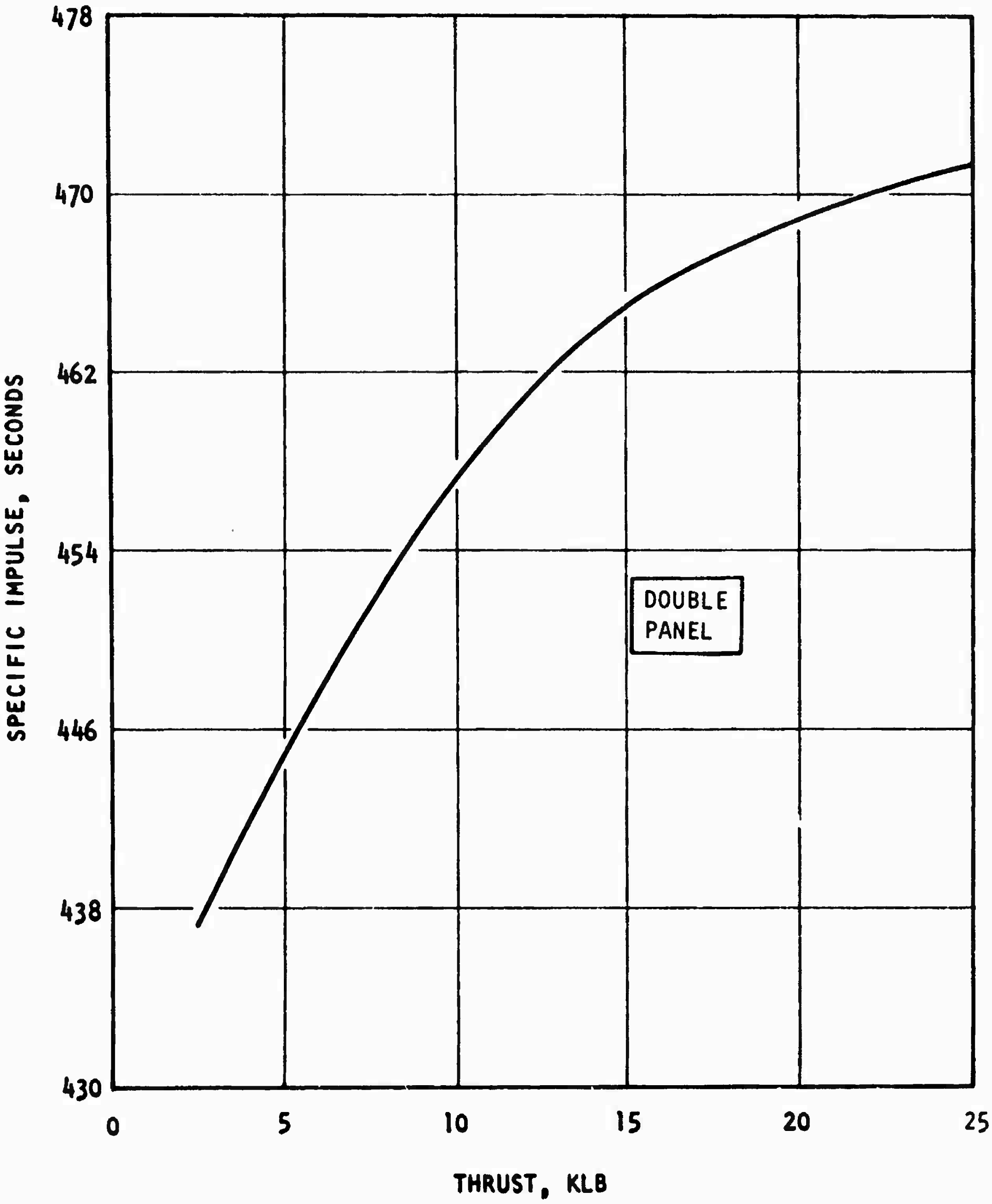


Figure 137. Engine Performance for Throttled Thrust Levels

Cooling Requirements

The most restrictive operating conditions with respect to thrust chamber heat transfer are not affected by throttling requirements.

Start Transient

The start transient is unaffected by elimination of the throttling requirement. The closed-loop control system would be retained to minimize the effects due to variations in the start conditions, i.e., tank pressures and hardware temperatures.

Throttling capability of 10:1 would necessitate a change in the start sequence, but the effects on the transient operating conditions are not significant. At 1.5 and 2.0 seconds into the start transient for the baseline engine, the turbine bypass and oxidizer turbine valves, respectively, are ramped open to predetermined positions to produce a thrust of 5000 pounds, i.e., throttled 5:1. Engine operation is turned over to closed-loop control 1 second later to enable start to the desired thrust level. With increased throttling capability, the only change in the start sequence is opening of the two control valves at 1.5 and 2.0 seconds to produce 2500 pounds thrust (10:1 throttled) instead of 5000 pounds.

Complexity

Complexity of the engine system is not reduced significantly by eliminating the throttling requirement because the control system used for throttling is required for start and mixture ratio control. Consequently, the only change is removal of control system circuitry associated with acceptance of throttling commands by the engine and generation of signals to direct movement of the control valves as a function of desired thrust level. Elimination of the throttling requirement would preclude any concern associated with turbopump critical speed characteristics because the turbopumps would not be required to operate over a range of steady-state speeds.

Throttling to 10 percent of design thrust has been evaluated with the off-design balance model. The operating ranges of the control valves and the injector flow characteristics are within normal design practice to provide sufficient control capability and prevent combustion instability. The increased operating range of the turbopumps will require added attention, however, to investigate critical speed characteristics.

Technology Improvements Needed

No improvements in technology are required for the range of throttling capability considered in this study.

Demonstrator Engine Development Schedule and Costs

Though the effect of eliminating throttle capability on engine design is insignificant, the demonstrator program cost is reduced. This reduction is due to the elimination of throttling-related test activity. The program duration is consequently slightly shorter and the costs associated with test operations, test consumables, and engineering support to test are reduced. Increasing throttling capability from the nominal 5:1 to 10:1 increases these test-related costs because the throttling-related test activity is increased. The somewhat larger turbomachinery needed for the 10:1 throttling also results in added cost. The change in program cost from 5:1 to 10:1 throttling is about 70 percent due to the difference in test requirements and about 30 percent due to the difference in turbomachinery size-related costs. The effects of throttling on cost are presented in Table 67.

Engineering Development Schedule and Costs

The engineering development program cost variations with throttling capability also are presented in Table 67 . The same factors which cause the demonstrator program cost changes also are responsible for the variations in development program cost.

TABLE 67. EFFECT OF THROTTLING CAPABILITY ON DOUBLE-PANEL DEVELOPMENT SCHEDULE COSTS

	Throttling Capability		
	None	5:1 (Nom.)	10:1
Demonstrator Program Cost, $ Δ Cost, $	27.4 M - .8 M	28.2 M 0	29.0 M + .8 M
Demonstrator Program Length, months Δ Length, months	30.5 - .5	31 0	31.4 + .4
Development Program Cost, $ Δ Cost, $	101.7 M -3.0 M	104.7 M 0	107.4 M +2.7 M
Development Program Length, months Δ Length, months	53.2 -1.8	55 0	56.2 +1.2
F[illegible]st Unit Cost, $ Δ Cost, $	825 K 0	825 K 0	839 K +14 K

First Production Unit Cost

The elimination of throttling capability does not change the first unit cost from its nominal value of 825 K. The larger turbomachinery required for the 10:1 throttling engine increases the cost 14 K over the nominal 5:1 throttling engine.

IDLE-MODE CAPABILITY

The baseline engine system is not designed for idle mode operation where both propellants flow directly to the thrust chamber under tank head without rotation of the turbopumps. The design changes required to incorporate this capability into the engine and the performance during idle mode are summarized.

Specific Impulse

Because the injector and thrust chamber are operating at conditions extremely different from design during tank-head idle mode, an accurate prediction of performance is not appropriate without test data. Delivered specific impulse, based on extrapolations of analytical predictions, is estimated to be approximately 430 seconds.

Weight

Tank-head idle mode requires addition of an on/off, hot-gas valve at the inlet of the fuel turbine to prevent rotation of the fuel turbopump. This valve weighs approximately 10 pounds. If thrust levels which are higher than can be achieved

with tank-head propellants are desired or, if the engine mixture ratio is unacceptably low, the driving pressure of the fuel or both propellants must be increased. If this increase were accomplished by allowing rotation of the fuel turbopump, the additional hot-gas valve would not be necessary.

Interface Dimensions

Addition of idle-mode capability has no effect on interface dimensions.

Cooling Requirements

Thrust chamber cooling reqirements necessitate a reduction in the engine mixture ratio to 3.9 during tank-head idle-mode operation. The resulting coolant bulk temperature is 1860 R at the jacket exit. The heat transfer limits are shown in Fig. 138.

Engine System Operating Conditions

With oxygen and hydrogen tank pressures of 40 and 20 psia, respectively, the nozzle stagnation pressure is approximately 8 psia. Thrust is approximately 180 pounds. If higher thrust levels are desired or if a mixture ratio of 3.9 is unacceptably low, the driving pressure of the fuel or both propellants must be increased to increase the nozzle stagnation pressure. This can be accomplished by increasing tank pressures or allowing turbopump rotation. Operation of the turbopumps should be avoided, however, because flow oscillations are likely to occur with the pumps throttled to the operating conditions required for idle-mode thrusts.

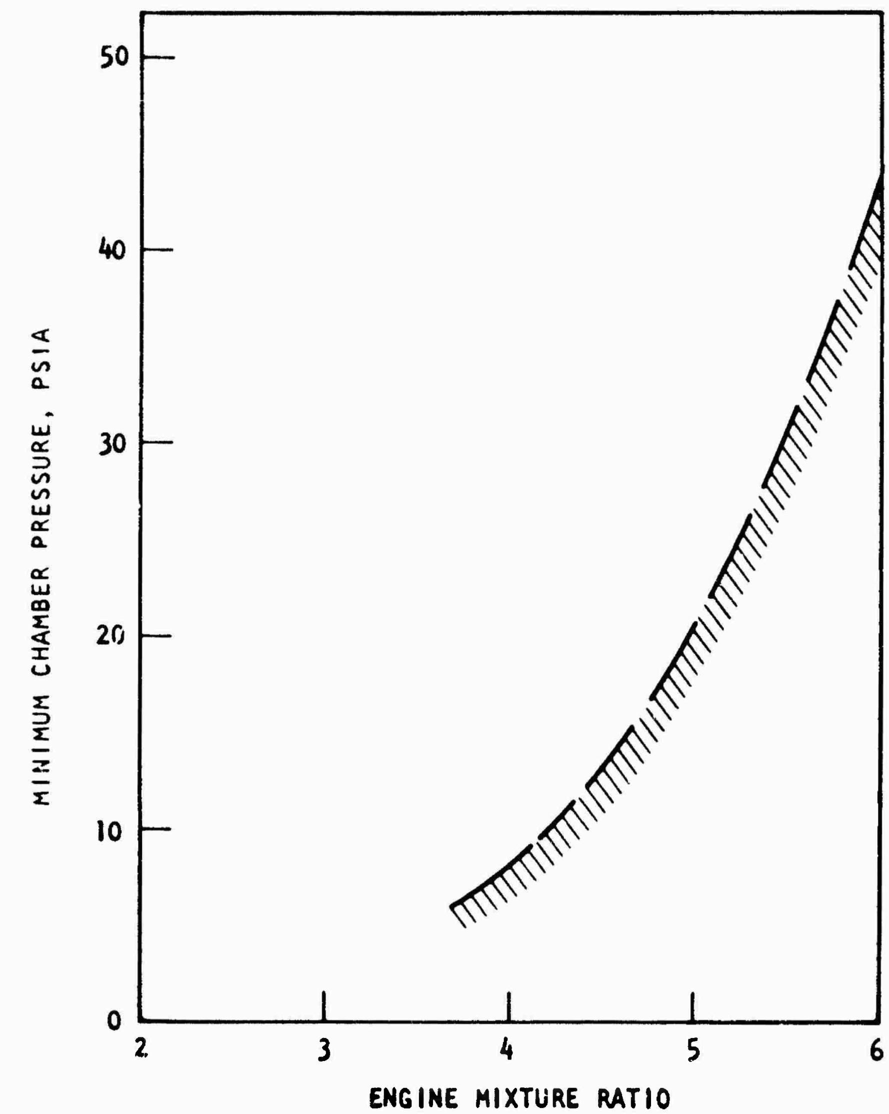


Figure 138. Tank-Head Idle-Mode Heat Transfer Limiting Minimum Chamber Pressure for Double-Panel Engine

Start Transient

Tank-head idle-mode capability requires an on-off fuel turbine inlet valve to prevent fuel turbopump rotation and a two-position oxidizer main valve to control mixture ratio. If the main oxidizer valve is not adequate in controlling the small oxidizer flow to the desired accuracy, a small-diameter line and valve in parallel with the main valve will be utilized during idle-mode operation. Sequencing of the control valves must be altered from that of the baseline engine to incorporate idle-mode operation into the start sequence because start to any thrust level from idle-mode operation was assumed.

The modified start sequence is shown in Fig. 139. The main fuel and turbine bypass valves are ramped full open to establish fuel coolant flow. Both turbine inlet valves remain closed to prevent turbopump rotation. (In the baseline engine start transient, the turbine bypass valve remains closed for 1.5 seconds and, without a fuel turbine inlet valve, the fuel turbopump is powered by the gaseous coolant flow.) The main oxidizer valve is then opened to its intermediate position and idle-mode operation continues for the desired duration. The engine can then either be shut down or started to a thrust level between 5000 and 25,000 pounds.

Figure 139 also shows a start to full thrust. By opening the fuel turbine inlet valve, and closing the turbine bypass valve until the fuel turbopump is powered to 50 percent of design speed, enough fuel flow is generated to cause the engine to bootstrap to the desired thrust. The main oxidizer valve is then ramped full open and the oxidizer turbine inlet valve is partially opened to allow oxidizer turbopump rotation. Approximately one second later, the turbine bypass and oxidizer turbine inlet valves are turned over to closed loop control for the remainder of the start transient. The start time, exclusive of extended idle-mode operation, is not significantly affected.

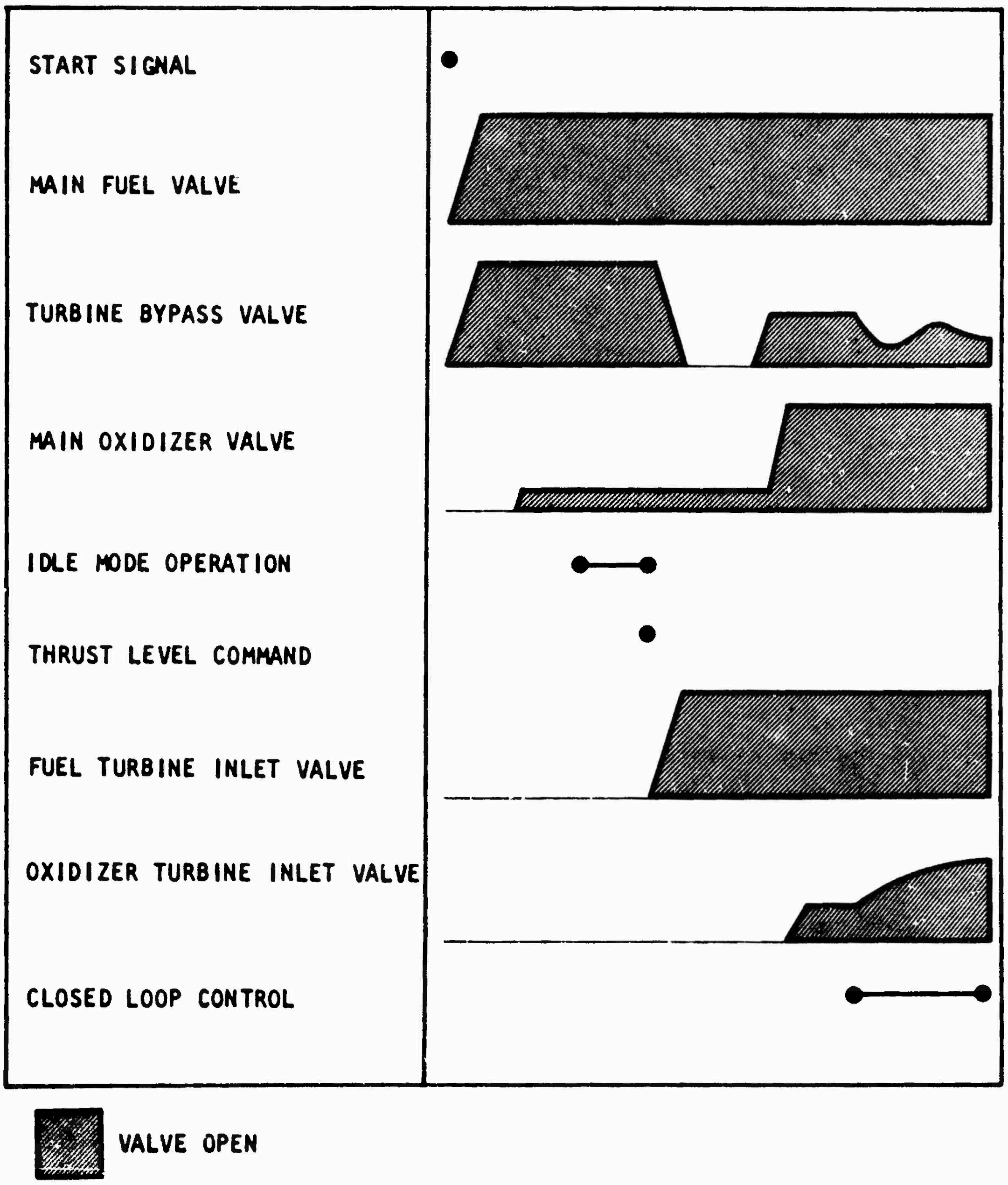


Figure 139. Double-Panel Start Sequence Including Idle-Mode Operation

Complexity

Complexity of the engine system is not appreciably affected by adding the fuel turbine inlet valve or using a two-position main oxidizer valve. The possibility of combustion instability during pressure-fed operation, however, is a problem which, if preventative corrections are required, could lead to increased complexity of the thrust chamber and injector designs.

Technology Improvements Needed

No improvements in technology are required to provide idle-mode capability. Engine design and development will utilize J-2S idle-mode experience (NAS8-19).

Demonstrator Engine Program

The engine modifications for idle mode and the effort expended to develop and demonstrate this capability add to the cost of the demonstrator program. The principal cost areas which increase are the valve program because of the new fuel turbine inlet valve and adding two-position control to the main oxidizer valve; the engine test program hardware because the engines now have the added valve costs; and the test operations, test consumables, and engineering test support for the idle-mode-related test activity in the thrust chamber and engine system programs. The cost and program duration effect of idle mode are presented in Table 68.

Development Program Schedule and Costs

Included in Table 68 with the demonstrator cost effect is the development program cost effect. The same areas of cost impact were responsible for the development program cost increase that caused the demonstrator program cost change.

First Unit Cost

The new fuel turbine inlet valve and the addition of two-position capability to the main oxidizer valve adds 30 K dollars to the cost of the first production unit.

TABLE 68. EFFECT OF IDLE MODE ON DOUBLE-PANEL DEVELOPMENT SCHEDULE COSTS

	Nominal Engine	Idle-Mode Engine
Demonstrator Program Cost, $	28.2 M	29.6 M
Δ Cost, $	-0-	+ 1.4 M
Demonstrator Program Length, months	31	31.6
Δ Length, months	-0-	+ .6
Development Program Cost, $	104.7 M	109.4 M
Δ Cost, $	-0-	+ 4.7 M
Development Program Length, months	55	57
Δ Length, months	-0-	+ 2
First Unit Cost, $	825 K	855 K
Δ Cost, $	-0-	+ 30 K

ANALYSIS OF SINGLE-PANEL ENGINE

VARIATION OF DESIGN ENGINE MIXTURE RATIO

The baseline engine system has a design mixture ratio of 5.5. The effects of varying this design specification over a range from 5 to 7 are summarized.

Nozzle Area Ratio

The chamber pressure (750 psia) and nozzle area ratio (110) of the baseline engine were chosen to correspond to the demonstrator thrust chamber configuration. Because the baseline area ratio was not selected on the basis of the design optimization study, area ratio is not increased to the heat ransfer limit as the design mixture ratio is varied. However, as mixture ratio is increased from 6.6 to 7.0, the heat transfer limit is exceeded and the area ratio must be lowered to 100. This variation of area ratio is shown in Fig. 140.

Specific Impulse

The effect of varying mixture ratio and the associated change in nozzle area ratio on delivered specific impulse is shown in Fig. 141. Engine performance decreases as mixture ratio increases.

Weight

The effect of varying mixture ratio on engine system dry weight is shown in Fig. 142. Engine weight decreases as mixture ratio increases, principally because of variations in the weights of turbopumps and ducts. Between mixture ratios of 6.6 and 7.0, the effect of nozzle area ratio on engine diameter contributes to the weight variation shown.

Envelope Dimensions

The effects of varying mixture ratio on engine envelope dimensions are shown in Fig. 143. The variations in length and diameter are due to changes in the nozzle area ratio.

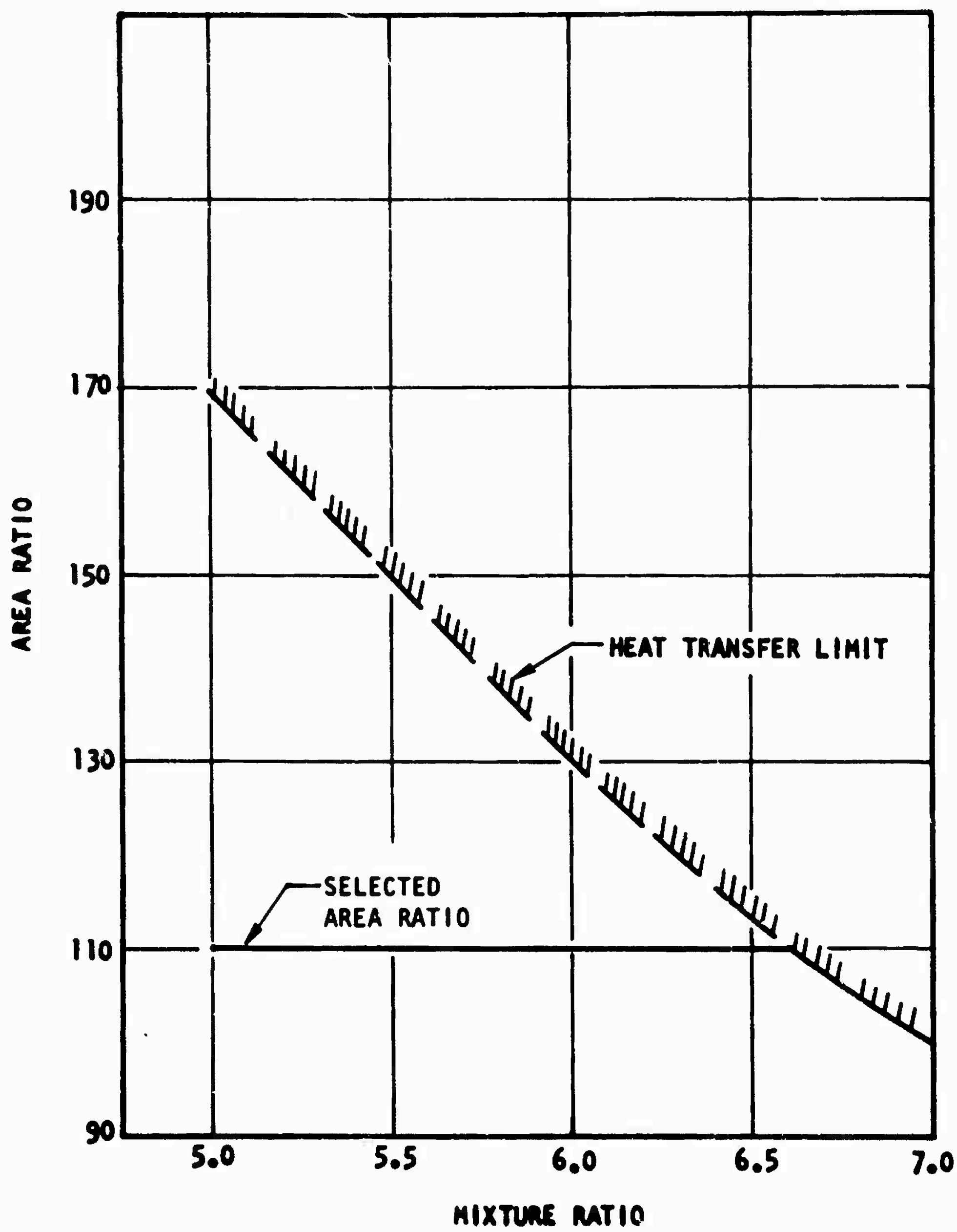


Figure 140. Effect of Single-Panel Design Engine Mixture Ratio on Nozzle Area Ratio

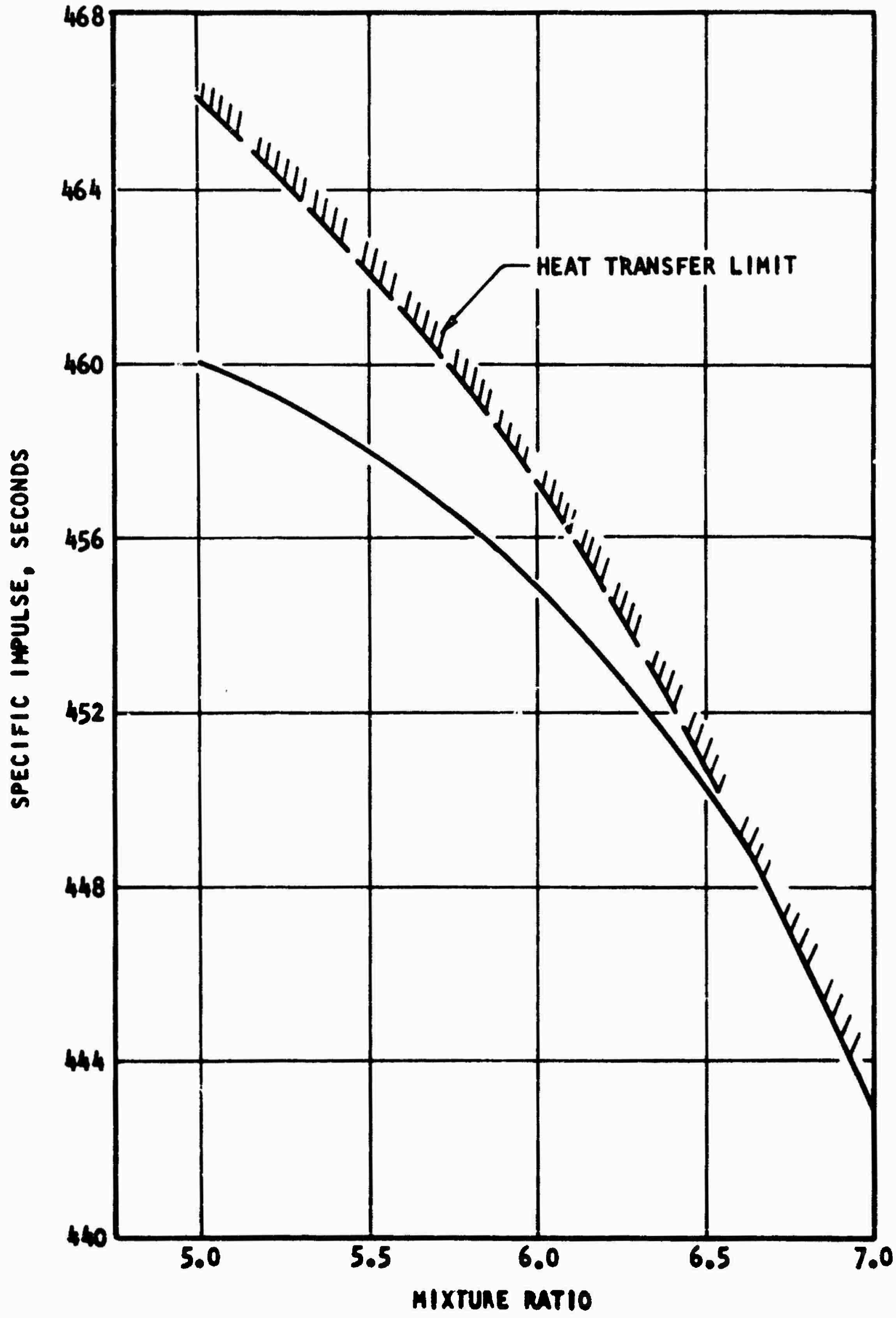


Figure 141. Effect of Single-Panel Design Engine Mixture Ratio on Delivered Vacuum Engine Specific Impulse

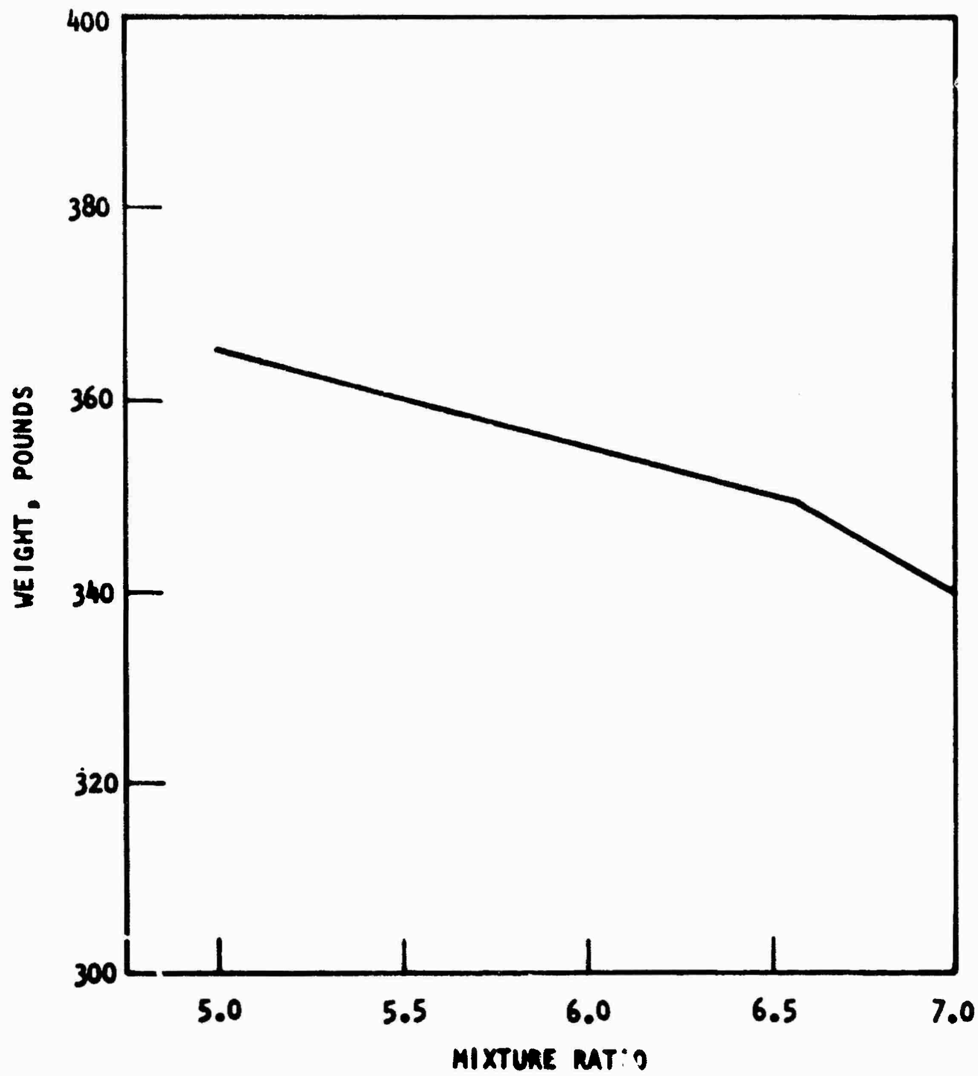


Figure 142. Effect of Single-Panel Design Engine Mixture Ratio on Engine Dry Weight

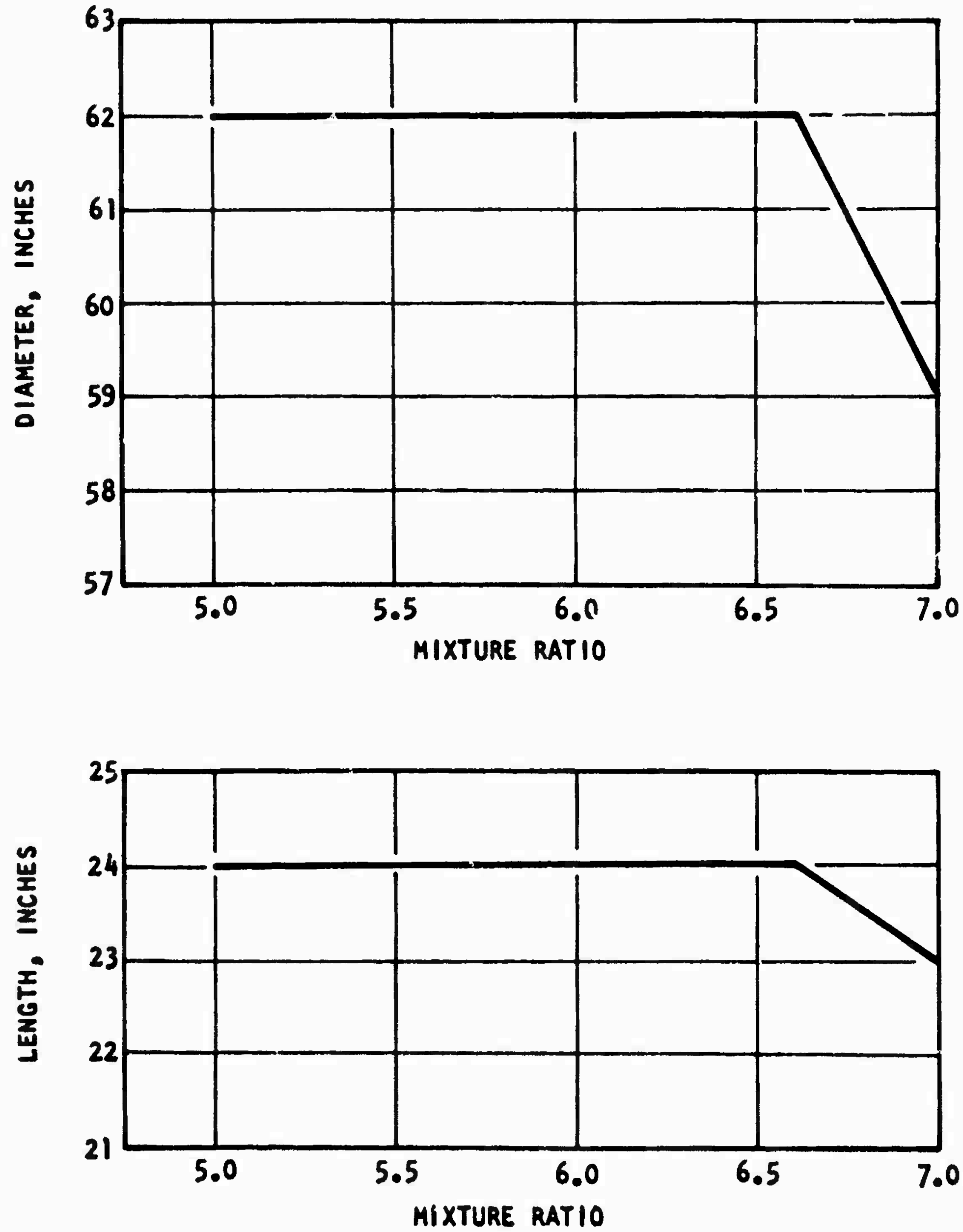


Figure 143. Effect of Single-Panel Design Engine Mixture Ratio on Engine Envelope Dimensions

Interface Dimensions

Variation of engine mixture ratio over a range from 5 to 7 does not affect interface dimensions. The inside diameter of the main fuel valve inlets is 2.5 inches and the corresponding diameter of the oxidizer valve is 3.0 inches. Except for inlet and exit dimensions, the values are of common design, however.

Cooling Requirements

For a constant area ratio of 110, the jacket exit temperature increases as mixture ratio is increased from 5.0 to 6.6. Thrust chamber cooling requirements are not affected by variations in the design mixture ratio between 6.6 and 7.0 because the nozzle area ratio was varied to maintain a constant maximum hot-gas wall temperature. Jacket exit temperature is shown as a function of mixture ratio in Fig. 144.

Engine System Operating Conditions

The effects of varying the design mixture ratio on turbomachinery parameters are shown in Fig. 145. The turbine bypass flow was maintained at 20 percent of the total fuel flow.

Technology Improvements Needed

No technology improvements are needed for variations in the design mixture ratio. Variation of the nozzle area to meet life specifications results in operating conditions which do not exceed the design criteria established for the baseline engine.

Demonstrator Engine Program Schedule and Costs

The demonstrator engine program cost decreases with increasing mixture ratio as indicated in Fig. 146. Three factors contribute to this trend. First, the turbomachinery fabrication and engineering costs decrease for the reduced overall turbomachinery size. Second, increasing mixture ratio decreases the requirement for the relatively expensive hydrogen while increasing the amount

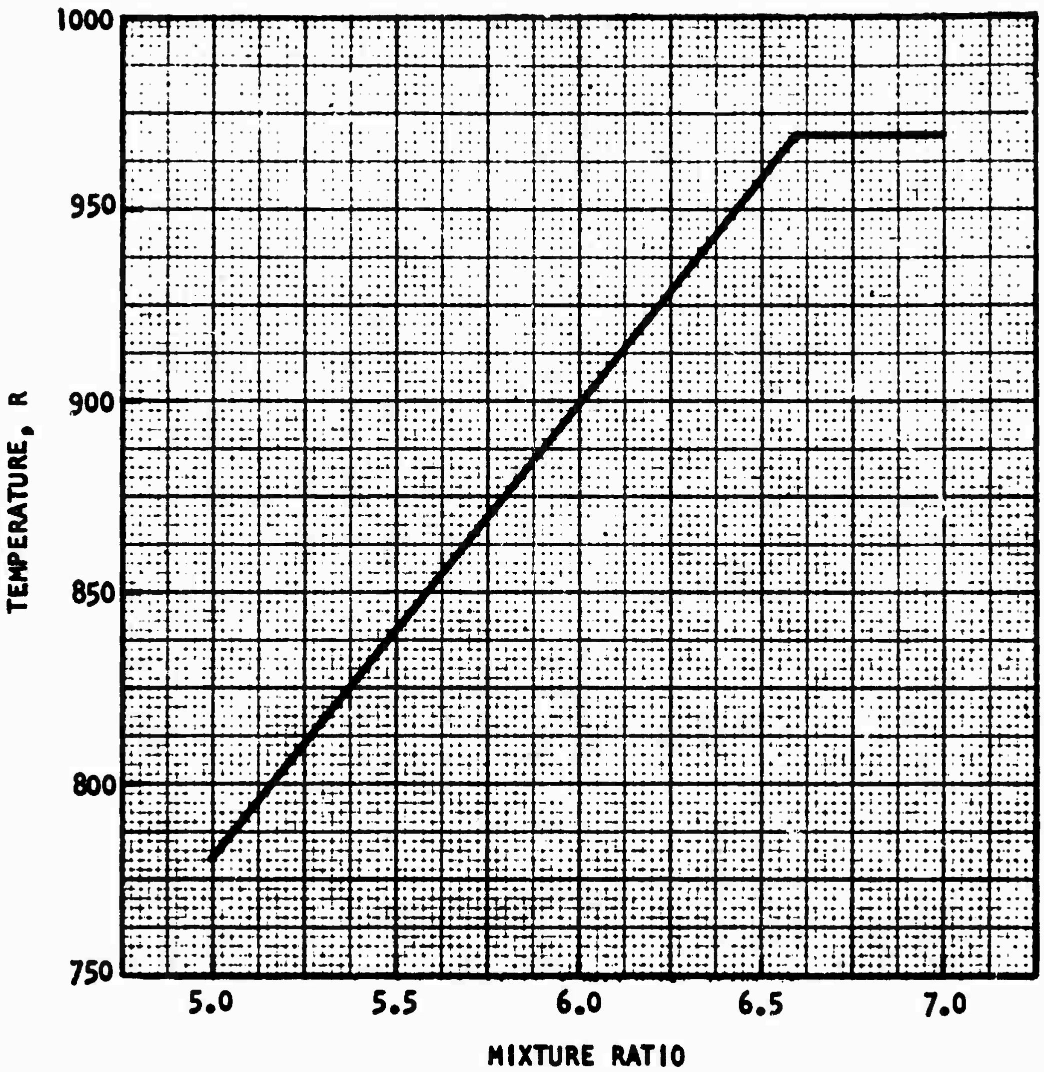


Figure 144. Effect of Single-Panel Design Engine Mixture Ratio on Jacket Exit Temperature

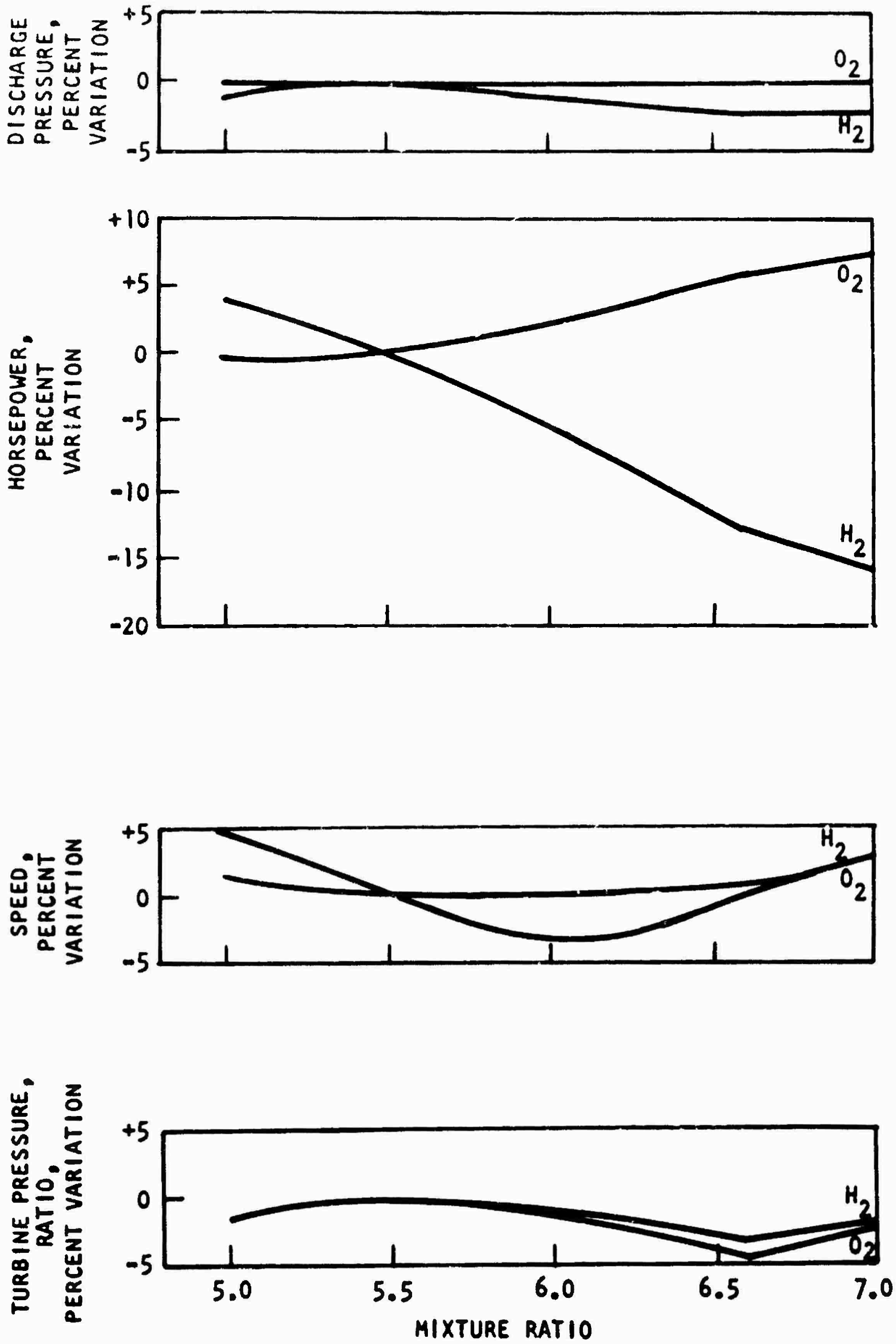


Figure 145. Effect of Single-Panel Design Engine Mixture Ratio on Turbomachinery Parameters

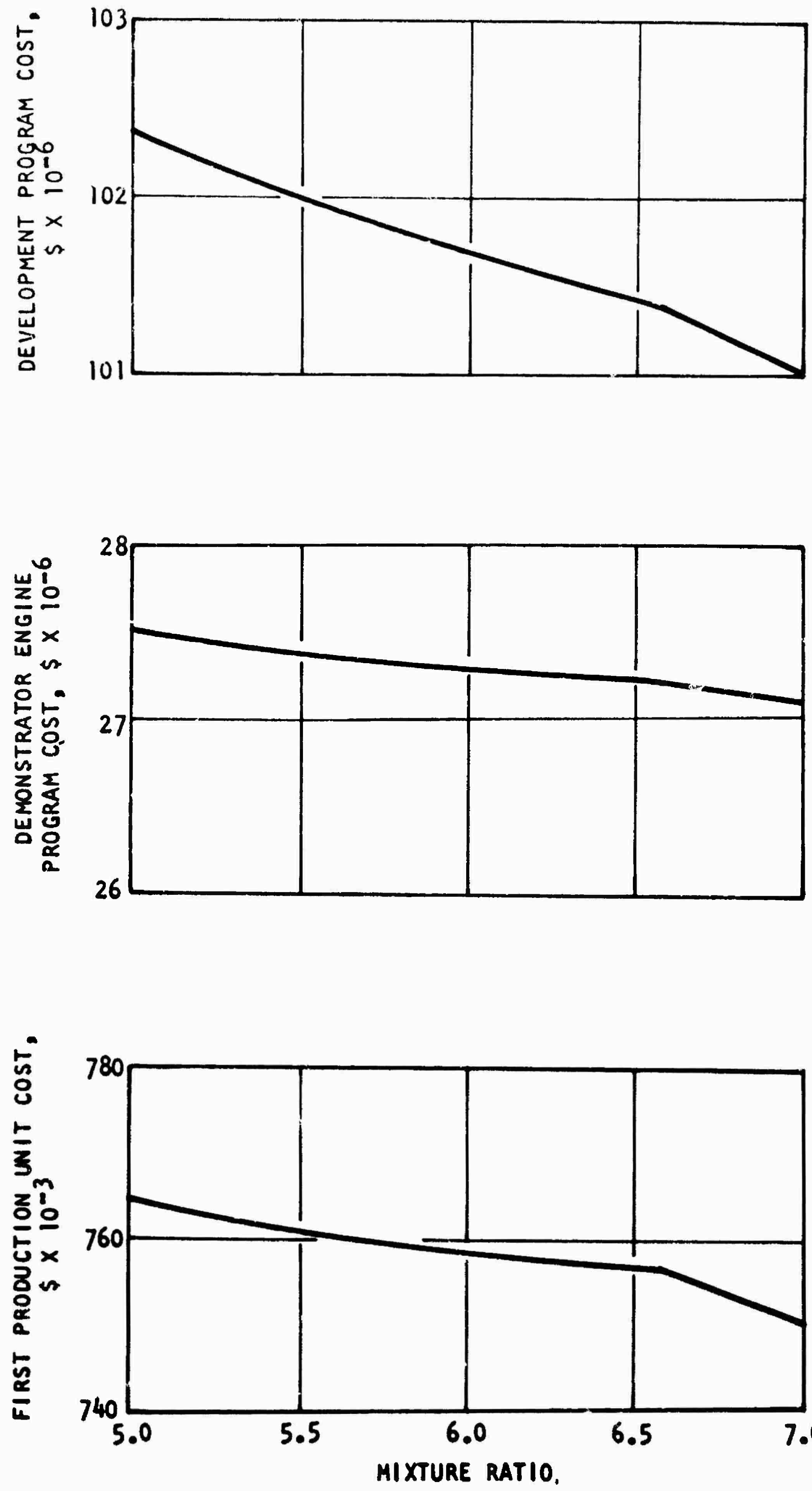


Figure 146. Effect of Single-Panel Design Engine Mixture on Development Schedule and Costs

of the cheaper oxygen. The total propellant cost, therefore, decreases as mixture ratio is increased. The final effect is the decrease in area ratio which occurs at mixture ratios above 6.6 and reduces the thrust chamber assembly fabrication and engineering costs.

The component differences resulting from the mixture ratio variation will not significantly affect the design or fabrication times required. The types of tests required and test frequencies will not be affected by mixture ratio. Consequently, no change in the demonstrator engine program schedule occurs with the mixture ratio variation.

Development Program Schedule and Costs

The development program cost will decrease with increasing mixture ratio as indicated in Fig. 146. As in the demonstrator engine program, the reduced turbomachinery size, the lower propellant cost and, at mixture ratios above 6.6, the decreasing thrust chamber size are responsible for this trend. Also, as in the demonstrator engine program, no change in the program schedule is required.

First Production Unit Cost

Figure146 includes the first unit cost trend with mixture ratio. The reduced total turbomachinery weight with increasing mixture ratio and the decrease in thrust chamber area ratio at mixture ratios above 6.6 result in a reduction of the cost of fabricating the engine.

VARIATION OF DESIGN FUEL PUMP NPSH

The baseline fuel pump net positive suction head (NPSH) is 60 feet. The effects of changing this design specification to zero feet or in the range from 15 to 60 feet were determined and the results are presented.

A reduction in design NPSH to approximately 25 feet can be accomplished with relatively minor changes in turbopump design parameters. At a value of 25 feet, the power limit for 20-percent turbine bypass flow is reached. A further

reduction in design NPSH to approximately 20 feet can be attained by changing turbopump design parameters and reducing the bypass flow to 15 percent of the total fuel flow. A design NPSH below 20 feet requires a low-speed fuel low-pressure pump. Electric motor and hydrogen gas turbine drives were considered for the boost pump.

Weight

The effect of varying the design fuel pump NPSH on engine weight is shown in Table 69. As design NPSH is reduced from 60 to 25 feet, the engine weight increases. The weight increase is predominantly due to an increase in the size of the fuel pump, which results from a reduction in speed and an increase in horsepower. Engine weight does not vary significantly as NPSH is further reduced from 25 to 20 feet.

TABLE 69. EFFECT OF FUEL PUMP NPSH ON SINGLE-PANEL ENGINE WEIGHT

NPSH	60	25	20	15,0
Engine System Changes	-	Turbopump Design	Turbopump Design Bypass Flow (15%)	Turbopump Design Low-Pressure Pump
Weight	360	378	381	362

A fuel low-pressure pump is required for a design NPSH of less than 20 feet. The weight of the low-pressure pump and electric motor or hydrogen gas turbine drive is offset by the reduction in weight of the main fuel pump, which operates at a relatively high speed and low horsepower because of its high inlet NPSH. The slight weight increase from the nominal value is due to the larger hydrogen inlet valve.

Interface Dimensions

As the fuel pump NPSH is reduced from 60 to 20 feet, the inlet diameter of the main fuel valve increases from 2.5 to 2.9 inches. At lower values of NPSH, the valve inlet diameter is 3.1 inches. The oxidizer valve inlet diameter is unaffected (3.0 inches).

Engine System Operating Conditions

As the fuel pump design NPSH is reduced to 25 feet, a reduction in fuel pump speed is required, which lowers the efficiency and increases the fuel pump horsepower requirement. The increased horsepower requirement is met by altering the flow split to the parallel turbines and increasing the turbine pressure ratios. A comparison of design operating conditions as a function of fuel pump NPSH is presented in Table 70.

Further reductions in design NPSH can be achieved by increasing the available turbine power or reducing pump power requirements. The most practical method of increasing the available turbine power is to increase the turbine flow. Increasing the turbine flow (reducing turbine bypass flow to 15 percent) in conjunction with changes in turbopump design parameters, is capable of meeting the power requirements for an NPSH of approximately 20 feet without exceeding reasonable turbopump design criteria.

For a design NPSH of less than 20 feet, a low-speed fuel low-pressure pump must be used to allow a reasonable main fuel pump design. Although the optimum fuel turbopump design requires an NPSH of less than 150 feet a low-pressure pump head of approximately 370 feet must be developed to maintain low-pressure pump parameters within design limits. Because of these design limits, low-pressure pump parameters do not vary significantly as a function of inlet NPSH.

For a low-pressure pump head of approximately 370 feet, the horsepower requirement is so small as to make a gas-turbine drive impractical. Alternatives are the use of an electric motor drive or an increase in low-pressure pump head to 1250 feet and use of a hydrogen-driven gas turbine in parallel with the main turbines. Low-pressure pump parameters for an inlet NPSH of zero feet are shown in Table 71 for both types of drives. The low-pressure pump is designed for 25 percent vapor pumping capability. The choice between the low-pressure pumps does not significantly change the main fuel pump head requirement and, therefore, only one design for the main pump was presented in Table 70.

TABLE 70. EFFECT OF SINGLE - PANEL FUEL PUMP DESIGN NPSH ON TURBOPUMP OPERATING CONDITIONS

Fuel Pump NPSH, feet	60		25		20		15, 0	
Turbine Bypass Flow, percent	20		20		15		20	
Propellant	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2
Low-Pressure Pump	No	No	No	No	No	No	No	Yes
Main Turbopump Parameters*								
Discharge Pressure	0.0	0.0	0.0	+13.3	0.0	+8.5	0.0	-3.5
Horsepower	0.0	0.0	+1.1	+53.5	+1.0	+57.5	+1.7	-10.4
Speed	0.0	0.0	+1.6	-33.0	+0.4	-38.7	+6.2	+18.9
Turbine Pressure Ratio	0.0	0.0	+7.2	+18.0	+10.2	+11.6	-4.6	-4.7

*Values shown are percent variation from design value

TABLE 71. SINGLE-PANEL FUEL LOW-PRESSURE PUMP DESIGN PARAMETERS

Low-Pressure Pump Drive	Electric Motor	H_2 Gas Turbine
Head, feet	371	1250
Power, hp	8.6	32
Speed, rpm	25,800	38,800

Start Transient

Variations in the start time to 90 percent thrust as a function of fuel pump NPSH are expected to be small. The largest variation anticipated would occur with the use of a low-pressure pump if "time constants" associated with rotating inertias of the low-pressure pump and main fuel turbopump are significantly different from that of the baseline fuel turbopump. The variation would occur during the initial acceleration of the fuel turbopump under open-loop control. The maximum effect is expected to be ± 0.3 second for a baseline engine start time of 4.4 seconds.

Complexity

Engine system complexity is not affected until the design NPSH is reduced below 20 feet and a low-pressure pump is required. Addition of a low-pressure pump and either an electric motor drive or a turbine with associated gas flow duct in parallel with the main turbines increases system complexity. A gas turbine drive also may require an open-loop control valve to prevent overspeed during start if the low-pressure turbopump response is significantly faster than the main fuel turbopump.

Technology Improvements Needed

No technology improvements are needed for the hydrogen pump inlet conditions considered in this study. Satisfactory low-pressure and main pump designs can be achieved to meet inlet NPSH's of 15 to 60 feet with present inducer technology. For a zero NPSH specification, the low-pressure pump must be designed with two-phase pumping capability. Two-phase pumping has been investigated under two recent NASA contracts (Ref. 5 and 6). Analytical techniques have been developed to design and predict vapor pumping capability, and good agreement exists between these analyses and experimental results using hydrogen pumps with up to approximately 35 percent vapor, by volume, at the inlet. If higher capability were needed the state of the art could be advanced by investigating higher overall incidence-to-blade angle ratios.

Demonstrator Engine Program Schedule and Costs

As hydrogen pump NPSH is decreased from the nominal 60 feet, the required increase in the size of the hydrogen pump results in an increase in the turbomachinery-related engineering and the turbomachinery fabrication costs. The need for larger flow area main fuel valves also adds to the cost of these systems. For 15 feet of NPSH, there is also the increased complication of adding a hydrogen low-pressure pump which results in additional engineering and fabrication costs as well as increased pump test costs.

The hydrogen low-pressure pump design for 15 feet NPSH also is adequate for operation at zero feet NPSH. To obtain zero NPSH capability, however, the turbomachinery and engine system test program costs will increase because additional testing to evaluate and demonstrate vapor pumping capacity are required. For the testing, the two-phase propellant would be generated by a Frantz screen installed immediately upstream of the pump inlet. The vapor fractions can be determined from measurements of pressure and temperature in the pure liquid upstream of the screen and in the two-phase fluid downstream of the screen. This technique has been successfully used on previous programs without testing problems and has a relatively minor impact on facility costs. The additional engine system testing with vapor pumping increases the demonstrator engine program duration by 0.25 month when zero feet pump NPSH is required. The cost changes in the demonstrator engine program because of lower fuel pump NPSH values are presented in Table 72 .

Development Program Schedule and Costs

The cost increases in the development program which result from reduced hydrogen pump NPSH values also are presented in Table 72. The same factors which contributed to the cost increase in the demonstrator program also are responsible for the development program cost increase.

First Production Unit Cost

The fabrication cost increases resulting from the engine design changes necessary to meet lower hydrogen pump NPSH values are summarized in Table 72.

TABLE 72. EFFECT OF SINGLE-PANEL FUEL PUMP NPSH ON DEVELOPMENT SCHEDULE AND COSTS

	NPSH, feet				
	60 (nom)	25	20	15	0
Demonstrator Program Cost, dollars	27.4 M	27.7 M	27.8 M	27.8 M	28.2 M
ΔCost, dollars	0	+ .3 M	+ .4 M	+ .4 M	+ .8 M
Demonstrator Program Length, months	31	31	31	31	31.25
ΔLength, months	0	0	0	0	+ .25
Development Program Cost, dollars	102.0 M	103.3 M	103.5 M	103.6 M	105.2 M
ΔCost, dollars	0	+1.3 M	+1.5 M	+1.6 M	+3.2 M
Development Program Length, months	55	55	55	55	56.4
ΔLength, months	0	0	0	0	+1.4
First Unit Cost, dollars	761 K	786 K	790 K	783 K	783 K
ΔCost, dollars	0	+25 K	+29 K	+22 K	+22 K

Variation of Design Oxidizer Pump NPSH

The baseline oxidizer pump design NPSH is 16 feet. The effects of varying this design specification to zero feet and over a range from 2 to 16 feet were determined and the results are presented.

A reduction in design oxidizer pump NPSH to approximately 4 feet can be accomplished by altering the turbopump designs. At a value of 4 feet, the oxidizer turbopump parameters approach design limits. Because of this, a design NPSH below 4 feet requires a low-speed oxidizer low-pressure pump. Electric motor and hydrogen gas turbine drives were considered for the low-pressure pump.

Weight

The effect of varying the design oxidizer pump NPSH on engine weight is shown in Table 73. As design NPSH is reduced from 16 to 4 feet, the weight increases primarily due to an increase in the sizes of the oxidizer pump and the oxidizer propellant inlet valve. The increase in the size of the pump results from a reduction in speed and an increase in horsepower.

TABLE 73. EFFECT OF SINGLE-PANEL OXIDIZER PUMP NPSH ON ENGINE WEIGHT

NPSH	16	4	2, 0
Engine System Changes	-	Turbopump Design	Turbopump Design Low-Pressure Pump
Weight	364	400	390

An oxidizer low-pressure pump is required for a design NPSH of less than 4 feet, but the increase in engine weight is primarily due to the larger oxidizer inlet valve. A portion of the weight of the low-pressure pump and electric motor or hydrogen gas turbine drive is offset by the reduction in weight of the main oxidizer pump which operates at a relatively high speed and low horsepower because of its high inlet NPSH.

Interface Dimensions

As the oxidizer pump NPSH is reduced from 16 to 11 feet, the inlet diameter of the main oxidizer valve is unchanged (3.0 inches). As NPSH is further reduced to 4 feet, the diameter increases to 3.5 inches. At lower values of NPSH, the oxidizer inlet diameter is 4.4 inches. The fuel valve inlet diameter is unaffected (2.5 inches).

Engine System Operating Conditions

As oxidizer pump design NPSH is reduced to 4 feet, a reduction in oxidizer pump speed is required, which lowers the efficiency and increases the required horsepower. The flow split to the parallel turbines also is altered, which increases the fuel turbine pressure ratio and fuel pump head. A comparison of design operating conditions for the baseline engine (oxidizer NPSH = 16 feet) and a system with an NPSH of 4 feet is presented in Table 74. The turbine bypass remains constant at 20 percent over the range of oxidizer NPSH values considered in this study.

TABLE 74. EFFECT OF SINGLE-PANEL OXIDIZER PUMP DESIGN NPSH ON TURBOPUMP OPERATING CONDITIONS

Oxidizer Pump NPSH, feet	16		4		2, 0	
Propellant	O_2	H_2	O_2	H_2	O_2	H_2
Low-Pressure Pump	No	No	No	No	Yes	No
Main Turbopump Parameters*						
Discharge Pressure, psia	0.0	0.0	0.0	+15.8	0.0	- 2.5
Horsepower, hp	0.0	0.0	+26.4	+13.6	- 4.7	- 7.5
Speed, rpm	0.0	0.0	-57.0	+18.5	+10.6	+16.7
Turbine Pressure Ratio	0.0	0.0	+21.4	+21.4	- 3.3	- 3.3

*Values shown are percent variation from design valve

For a design NPSH of less than 4 feet a low-speed oxidizer low-pressure pump must be used to maintain the main oxidizer pump design within established limits. A main pump NPSH of approximately 40 feet is necessary to achieve an optimum main pump design; a low-pressure pump head of 40 feet is required for an inlet NPSH of zero, and a correspondingly lower head must be developed for a design inlet NPSH of up to 4 feet. Low-pressure pump design parameters over this range of inlet NPSH do not vary significantly and only the zero NPSH design is presented in Table 75. The low-pressure pump is designed for 25-percent vapor pumping capability.

For an oxidizer low-pressure pump with a head of 40 feet or less, the horsepower requirement is so small as to make a gas-turbine drive impractical. Alternatives are the use of an electric motor drive or an increase in head of 125 feet and use of a hydrogen-driven gas turbine in parallel with the main turbines. The choice between low-pressure pumps does not significantly change the main oxidizer pump design because both provide at least the NPSH required for an optimum main pump design (40 feet).

TABLE 75. SINGLE-PANEL OXIDIZER LOW-PRESSURE PUMP DESIGN PARAMETERS

Boost Pump Drive	Electric Motor	H_2 Gas Turbine
Head, feet	40	125
Power, hp	6.8	18
Speed, rpm	4470	5960

Start Transient

The start transient is not significantly affected by variations in oxidizer pump NPSH for the range of values considered. The oxidizer turbopump acceleration is limited by the desired rate of increase in chamber pressure rather than the influence of NPSH on size and, therefore, rotating inertia of the pump.

Complexity

Engine system complexity is not affected until the design NPSH is reduced below 4 feet and an oxidizer low-pressure pump is required. Addition of a low-pressure pump and either an electric motor drive or a turbine with associated gas flow duct in parallel with the main turbines increases system complexity. A gas turbine drive also would require an open-loop control valve at the inlet of the oxidizer low-pressure pump turbine to prevent flow to this turbine during initial acceleration of the fuel turbopump.

Technology Improvements Needed

Technology is not now available for pumping two-phase oxygen, but a program is in progress to establish design criteria and test two-phase oxygen inducers (Contract NAS8-26645). Analytical techniques and test experience with two-phase hydrogen will be utilized in the program.

Demonstrator Engine Program Schedule and Costs

As oxygen pump NPSH is decreased from the nominal 16 feet to 4 feet, the required increase in the size of the oxygen pump results in an increase in the turbomachinery-related engineering and the turbomachinery fabrication costs. For the range of NPSH values between 4 and 2 feet, there also is the increased complication of adding an oxygen low-pressure pump which results in additional engineering and fabrication costs as well as increased pump test costs. The inlet diameter of the oxygen pump or low-pressure pump increases with decreasing NPSH which establishes a need for larger flow area main inlet valve which also adds to the cost of the lower oxygen pump NPSH systems.

The low-pressure pump design for an NPSH between 4 and 2 feet also is adequate for operation at zero feet NPSH. To obtain zero NPSH capability, however, the turbomachinery and engine system test program costs will increase because additional testing to evaluate and demonstrate vapor pumping capacity are required.

The cost changes in the demonstrator engine program because of lower oxidizer pump NPSH values are presented in Table 76.

Development Program Sechedule and Costs

The cost increases in the development program which result from reduced oxygen pump NPSH values also are presented in Table 76. The same factors which contributed to the cost increase in the demonstrator program also are responsible for the development program cost increases.

First Production Unit Cost

The fabrication cost increases resulting from the engine design changes necessary to meet lower oxygen pump NPSH values are summarized in Table 76.

TABLE 76. EFFECT OF SINGLE-PANEL OXIDIZER PUMP NPSH ON DEVELOPMENT SCHEDULE AND COSTS

	NPSH, feet			
	16 (nom)	4	2	0
Demonstrator Program Cost, dollars	27.4 M	28.1 M	28.4 M	28.8 M
Δ Cost, dollars	0	+ .7 M	+1.0 M	+1.4 M
Demonstrator Program Length, months	31	31	31	31.25
Δ Length, months	0	0	0	+ .25
Development Program Cost, dollars	102.0 M	104.5 M	105.4 M	107.1 M
Δ Cost, dollars	0	+2.5 M	+3.4 M	+5.1 M
Development Program Length, months	55	55	55	56.4
Δ Length, months	0	0	0	+1.4
First Unit Cost, dollars	761 K	810 K	820 K	820 K
Δ Cost, dollars	0	+49 K	+59 K	+59 K

THERMAL CYCLES (REUSABLE VERSION)

The baseline engine system (reusable version) is designed for 300 thermal cycles between overhauls. The effects of changing this design specification to 60 and 600 were investigated.

Nozzle Area Ratio

Because the baseline nozzle area ratio (110) was not selected to correspond to the heat transfer limit established in the engine design optimization study (ε = 150), the area ratio is not increased as the thermal cycle design requirement is lowered. Therefore, a reduction in the thermal cycle design specification to 60 does not have any impact on the engine system. Increasing the thermal cycle specification to 600, however, requires a reduction in the thrust chamber wall temperatures. The optimum mission performance criteria for these more restrictive heat transfer requirements results in a reduction in the nozzle area ratio, but the chamber pressure (750 psia) is not affected. The nozzle area ratio for the higher cycle life is 95.

Specific Impulse

Engine performance is not affected if the thermal cycle specification is reduced to 60. As the specification is increased from 300 to 600 cycles, specific impulse decreases from 458.0 to 455.5 seconds. The effect on delivered specific impulse is due to the variation in area ratio.

Engine Weight

The weight of the engine system is not affected if the thermal cycle requirement is reduced to 60. Engine weight decreases from 364 to 350 pounds as the life is increased from 300 to 600 cycles. The effect of cycle life on engine system dry weight is primarily due to the variation in area ratio, which directly influences the engine diameter.

Engine Cooling Requirements

Cooling requirements are unchanged for a lower cycle life specification. The higher cycle life capability is accomplished by reducing the temperature gradient from the hot-gas wall to the coolant channel closure wall. This requires a reduction in the hot gas wall temperature of approximately 65 R.

Engine Maintenance Requirements.

Engine maintenance requirements are not affected by variations in the number of thermal cycles over the range considered.

Engine System Operating Conditions

Engine system operating conditions are unchanged if the thermal cycle requirement is reduced. Variations in turbopump operating conditions for a 600-cycle specification are shown in Table 77.

TABLE 77. EFFECT OF INCREASING THE CYCLE LIFE REQUIREMENT ON SINGLE-PANEL TURBOPUMP OPERATING CONDITIONS

Thermal Cycles	300		600	
Propellant	O_2	H_2	O_2	H_2
Turbopump Parameters*				
Discharge Pressure	0.0	0.0	0.0	+0.1
Horsepower	0.0	0.0	+0.5	-1.8
Speed	0.0	0.0	+0.1	+6.1
Turbine Pressure Ratio	0.0	0.0	+1.9	+1.8

* Values shown are percent variations from design value

Technology Improvements Needed

No technology improvements are needed to provide a lower thermal cycle capability. However, to increase the engine thermal cycle capability, operating hot-gas wall temperatures must be reduced. The effect of the reduced wall temperature on other engine system and component operating conditions can be minimized if more detailed and exact thermal cycle fatigue and creep data were available for the specific materials that are to be used in the thrust chamber fabrication.

Demonstrator Engine Program Schedule and Costs

Thermal cycles will be accumulated by on/off cycling as part of specific tests. The number of cycles per test can be adjusted to account for the variations in thermal cycle design requirements without altering the number of tests required or the test costs. The difference in program cost is consequently affected only by the engineering and fabrication cost variations resulting from changes in the actual engine design. Because the engine design is unchanged by the reduction in thermal cycles to 60, the cost is unchanged. The reduction in area ratio necessary for the 600 thermal cycles reduces the demonstrator engine program cost by 0.1 million dollars, from the nominal 27.4 million dollars to 27.3 million dollars. The program schedule is unchanged by the thermal cycle variations.

Development Program Schedule and Costs

As in the demonstrator engine program, the only cost change results from the area ratio decrease at 600 thermal cycles. The cost decreases by 0.2 million dollars, from 102.0 million dollars to 101.8 million dollars. The program schedule is unchanged by the thermal cycle variation.

First Production Unit Cost

The 95 area ratio engine first production unit cost is 754 thousand dollars, lower by 7000 dollars than the nominal 110 area ratio engine cost of 761 thousand dollars.

THERMAL CYCLES (EXPENDABLE)

An increase in the number of required thermal cycles from 6 to 10 for the ground-based expendable version would not affect the engine system design, performance, or cost because the engine is designed for considerably more than 10 cycles for use in the development program.

NUMBER OF VACUUM STARTS

The nominal engine system is capable of 60 vacuum starts without requiring inspection or component servicing. A variation of this specification to 20 and 600 was investigated.

The engine system is designed to a maximum thrust chamber wall temperature which is consistent with the 10-hour life and the 300 thermal cycle specifications. Because lowering the vacuum start requirement does not affect either of these specifications, there is no resulting relaxation of restrictions on operating conditions. Therefore, a reduction of the requirement to 20 vacuum starts does not affect engine system or component design, performance, or cost.

Increasing the vacuum start capability to 600 necessarily implies an increase in the cycle life specification to 600 thermal cycles. The impact on the engine system is, therefore, the same as described in the section on Thermal Cycles (reusable version).

HOUR LIFE

The baseline engine system has a life specification of 10 hours between overhauls. Engine design is rather insensitive to variations in this requirement over a range from 2 to 20 hours, however, and performance and operating conditions are not affected.

Demonstrator Engine Program Schedule and Costs

Though the engine design is not affected by the life requirement, the life demonstration oriented testing is and, consequently, the costs for test operations, test consumables, and test support engineering change with engine life. This effect on the demonstrator engine program cost is presented in Fig. 147.

Development Program Schedule and Costs

The variation in test requirements with engine life for the development program are summarized in Fig. 148. These requirements establish the effect on development program length which is presented in Fig. 149. The cost as a function of life for the development program is included with the demonstration program cost on Fig. 147.

First Production Unit Cost

Because the engine design is insensitive to changes in the life requirement, the cost of fabricating the first production unit is constant with engine life.

MAXIMUM RUN TIME

The maximum run time specification is 1000 seconds. The effects on engine design and costs of changing this requirement to 500 and 2000 seconds are insignificant for fixed thermal cycle and hour-life requirements.

MAXIMUM ORBIT STORAGE TIME

The baseline engine system is designed for a maximum orbit storage time of 52 weeks. The primary consideration is protection against the possibility of vacuum cold welding, especially of the turbopump bearings. This protection will be provided by a regular rotation of the turbopumps or by pressurization of the engine feed system. Both techniques would utilize an inert gas from an external source. Pressurization of the feed system would require sealing of the thrust chamber

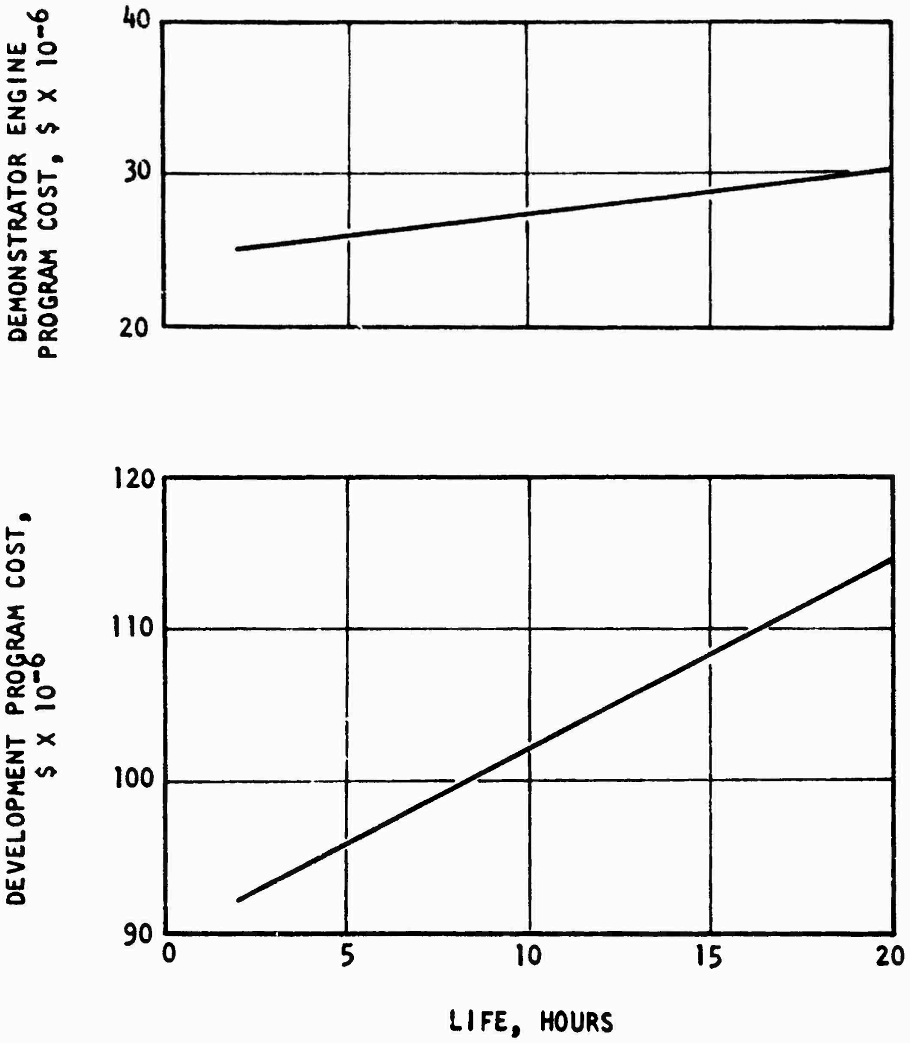


Figure 147. Effect of Life on Single-Panel Cost

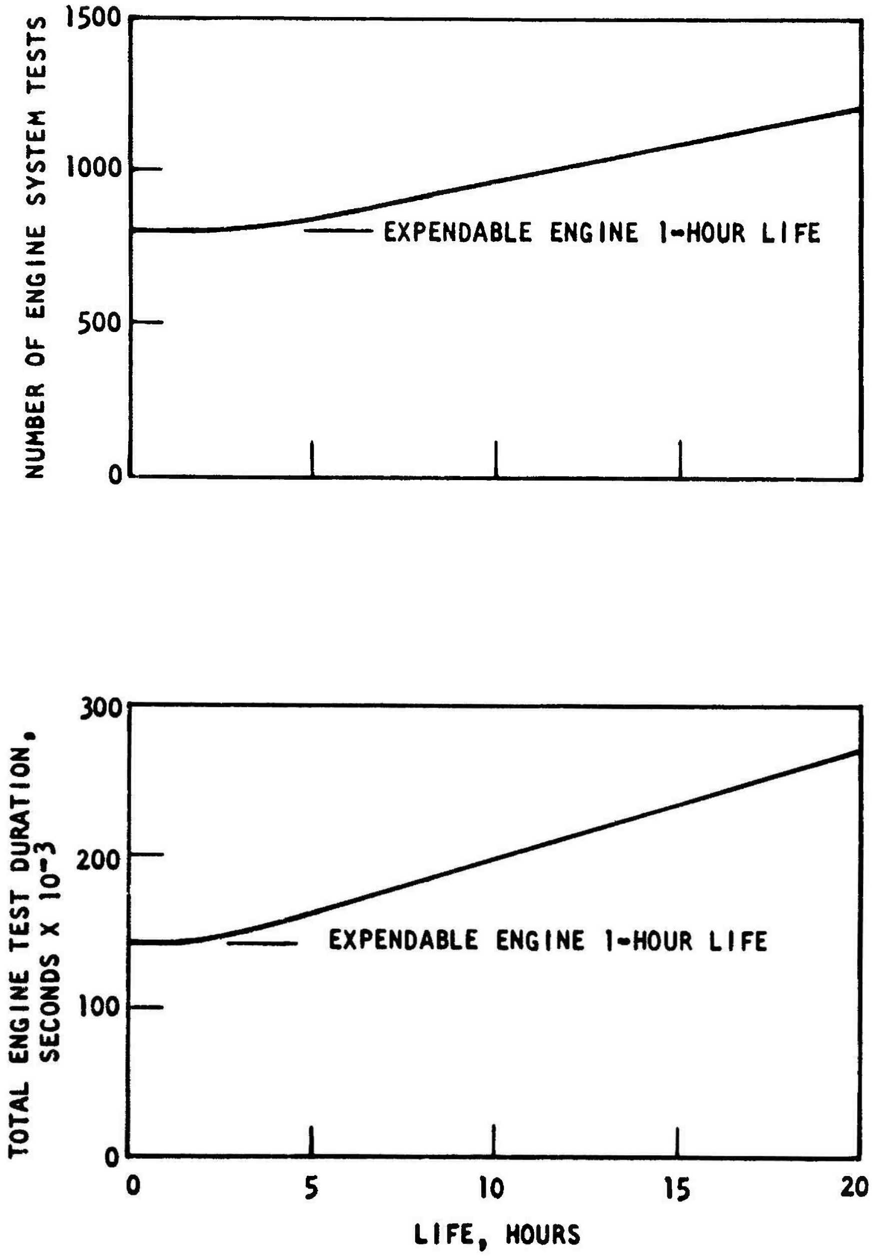


Figure 148. Effect of Life on Single-Panel Development Program Testing

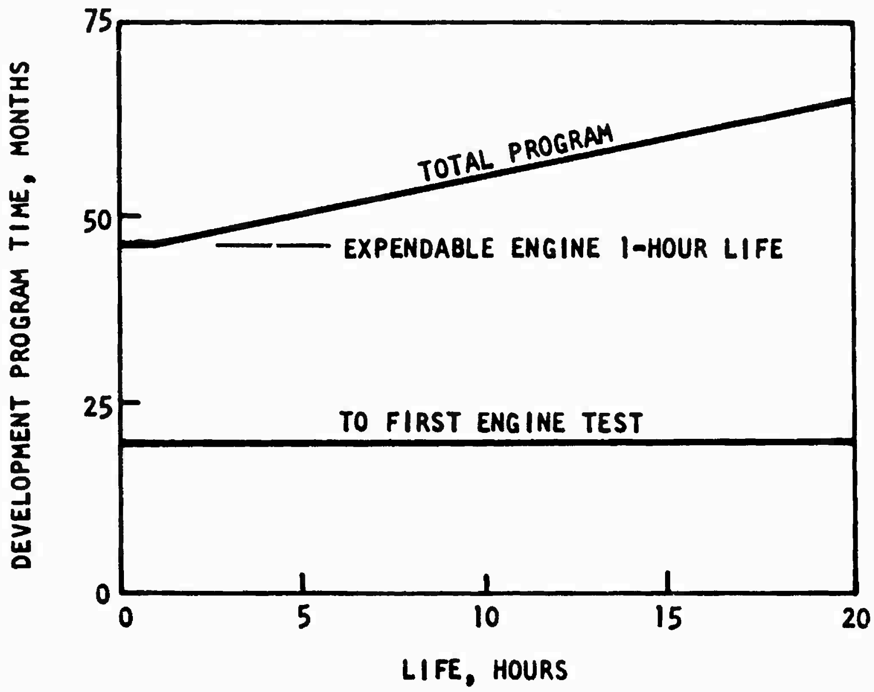


Figure 149. Effect of Life on Single-Panel Development Program Duration

throat. Based on the present knowledge of the vacuum cold-welding process, a variation in the maximum storage time specification over a range from 2 to 104 weeks does not significantly affect the engine system design or cost.

GIMBAL ANGLE

The aerospike engine design does not include flexible propellant supply ducting or gimbal actuators. The basic engine is, therefore, unchanged by gimbal angle variations between 3 and 7 degrees. There is no effect on weight, dimension, gimbal power requirement, or first production unit cost.

GIMBAL ACCELERATION

Weight

Engine system weight will increase by approximately 10 pounds when gimbal acceleration is increased from its nominal value of 5 rad/sec^2 up to 10 rad/sec^2 due to the increased gimbal mount strength required to withstand the increased actuator force and heavier mounts to support the engine components during the higher acceleration.

Interface Dimensions

Engine interface dimensions will not be affected by this variation.

Engine Maintenance Requirements

The engine components are capable of withstanding the increased accelerations without increasing their maintenance requirements.

Gimbal Power Requirements

Gimbal power is increased to 0.8 horsepower, an increase of 0.1 horsepower over the nominal engine requirements.

First Production Unit Cost

The strengthened gimbal and component mounts will have an insignificant effect on the first unit cost.

NOZZLE AREA RATIO

The baseline engine has a nozzle expansion area ratio of 110. The effects of varying this design specification between 100 and 400 was investigated.

Specific Impulse

The effect on vacuum delivered performance of varying nozzle area ratio from 100 to 150 is shown in Fig. 150. The area ratio cannot be increased above 150 because of heat transfer limits.

Weight

The effect of varying nozzle area ratio on engine system weight is shown in Fig. 151. The weight variation is due to the influence of area ratio on engine diameter.

Envelope Dimensions

Envelope dimensions increase as nozzle area ratio increases. Length and diameter data are shown in Fig. 152.

Cooling Requirements

The effect of increasing nozzle area ratio on jacket exit temperature is shown in Fig. 153.

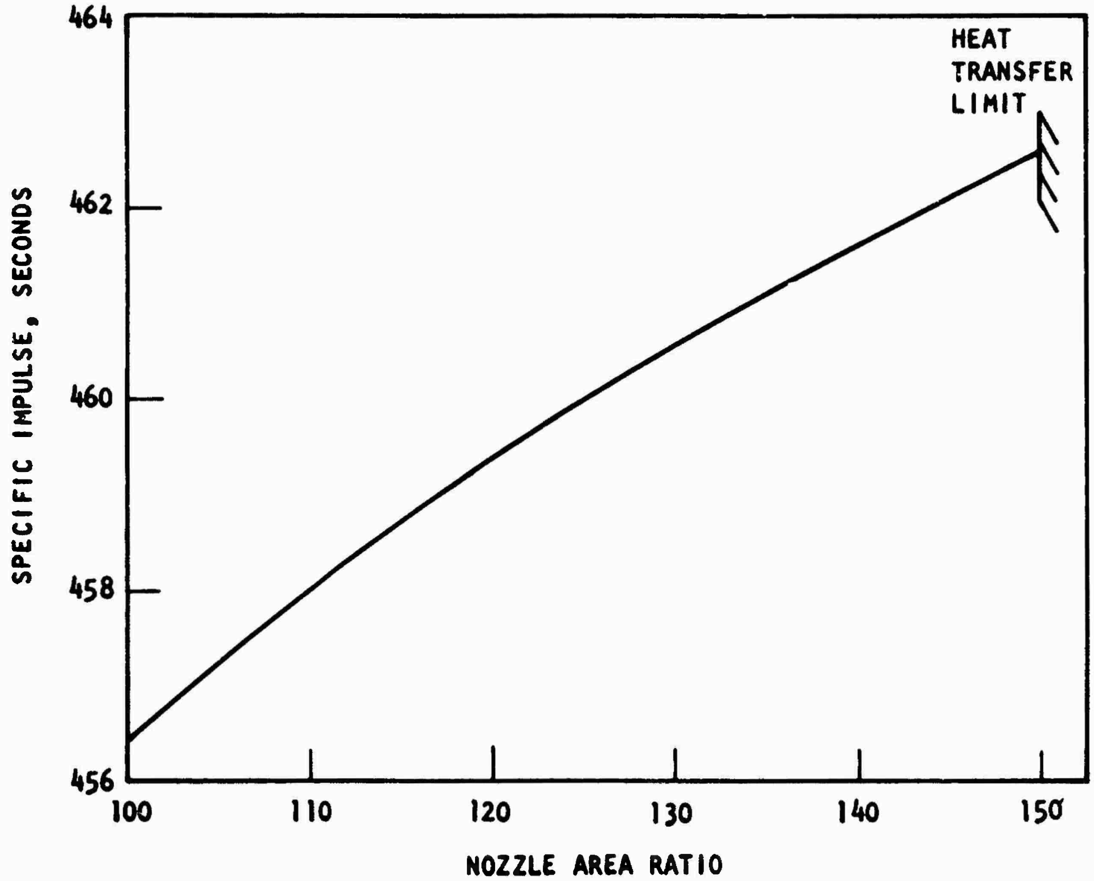


Figure 150. Effect of Single-Panel Nozzle Area Ratio on Delivered Vacuum Specific Impulse

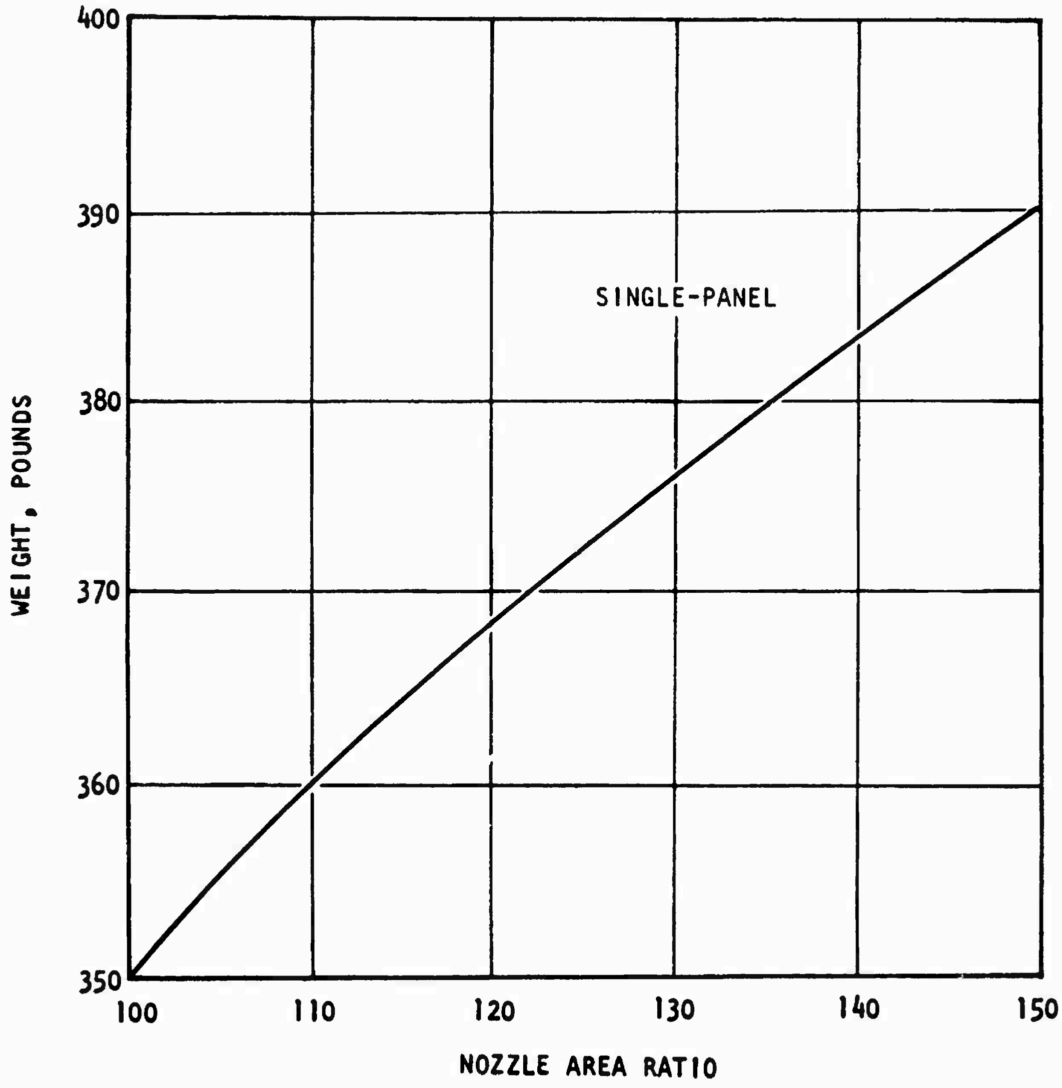


Figure 151. Effect of Nozzle Area Ratio On Engine Weight

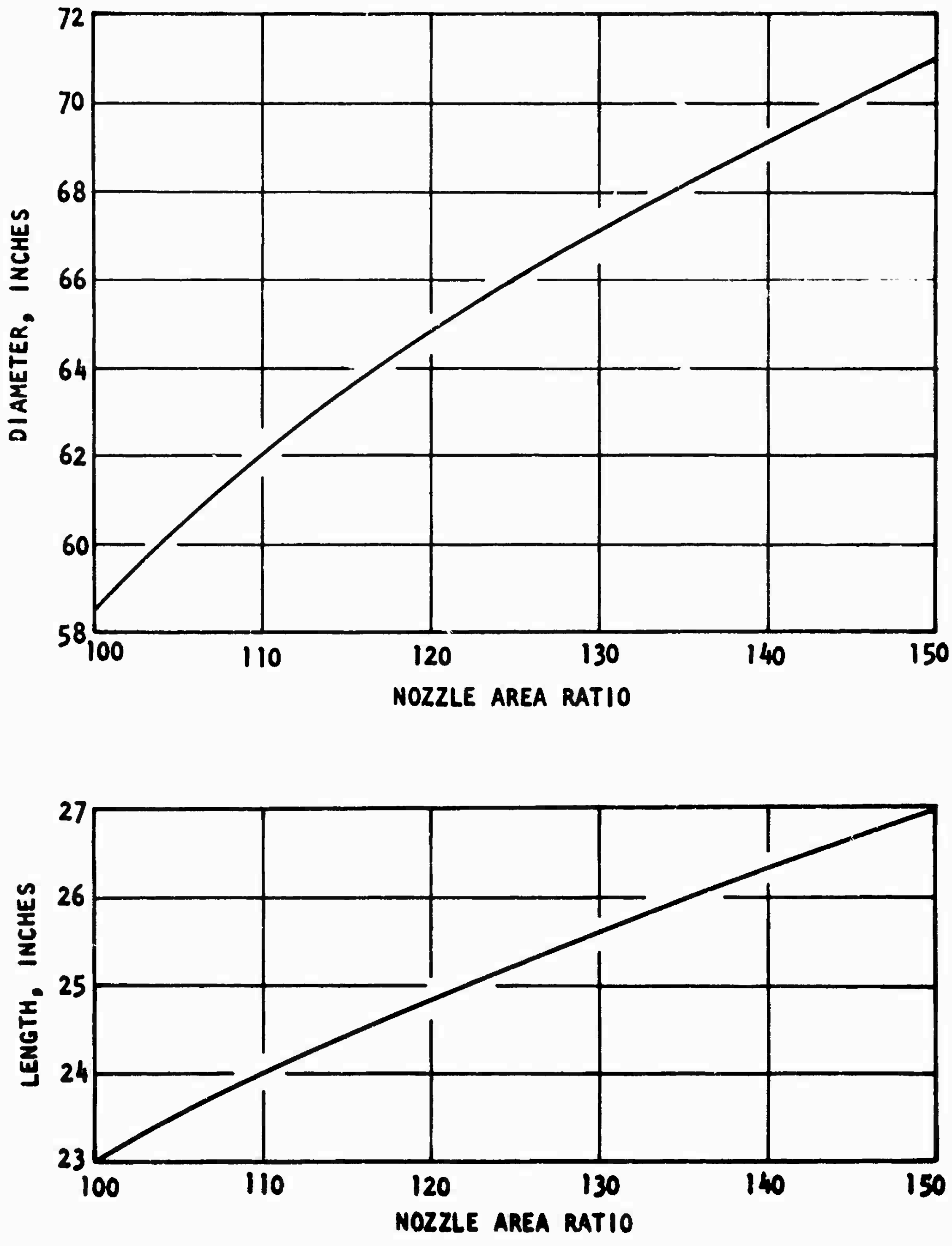


Figure 152. Effect of Single-Panel Nozzle Area Ratio on Engine Envelope Dimensions

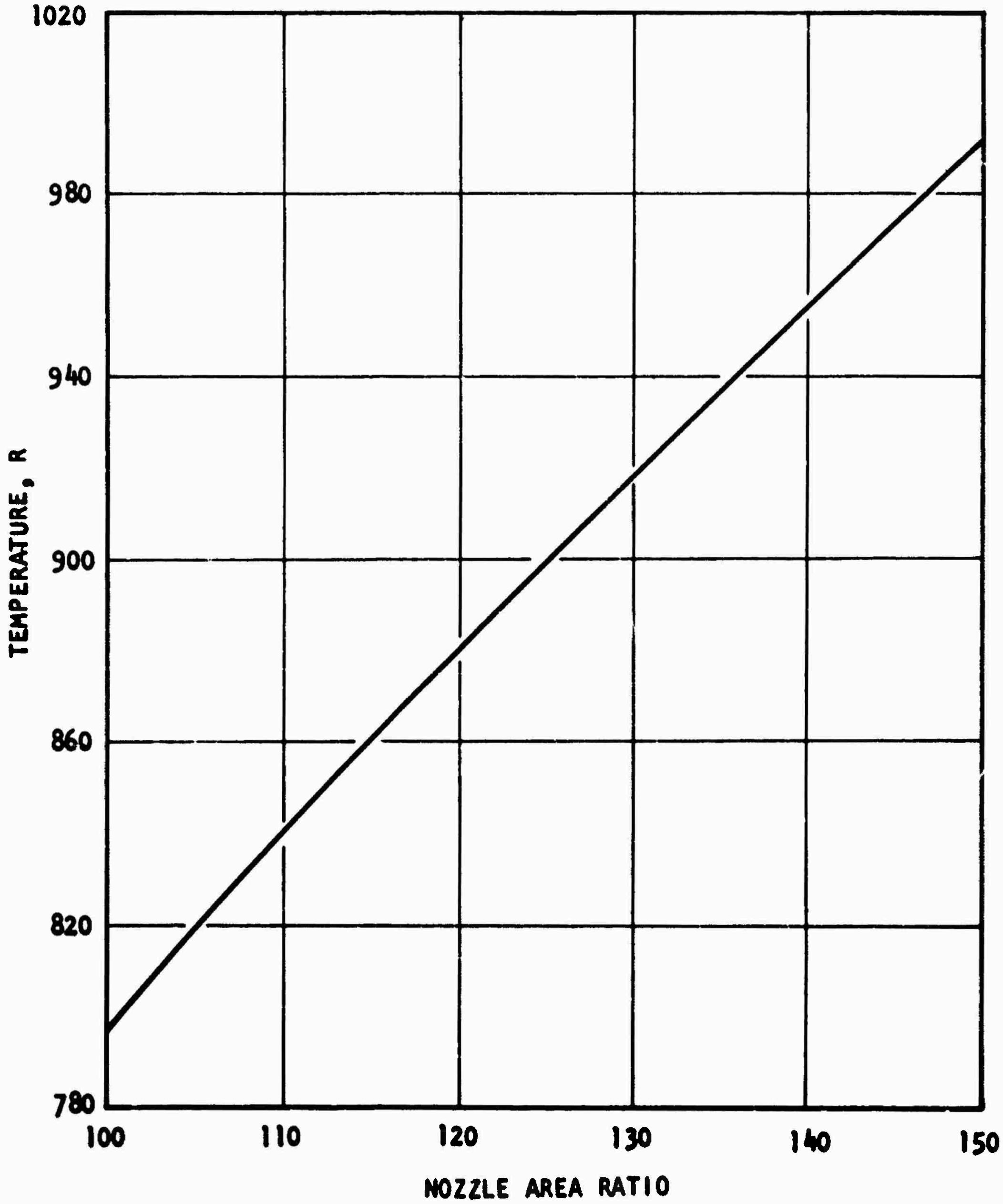


Figure 153. Effect of Single-Panel Nozzle Area Ratio on Jacket Exit Temperature

Gimbal Power Requirements

The gimbal power requirement varies from approximately 0.69 horsepower at 100 area ratio to 0.74 horsepower at 150 area ratio.

Technology Improvements Needed

No improvements in technology are needed for variations in nozzle area ratio between 100 and 150 because thrust chamber heat transfer is within established limits.

Demonstrator Engine Program Schedule and Costs

Due to the increasing size of the thrust chamber assembly as area ratio is increased, the costs associated with design functions such as stress analysis and fabrication of the thrust chamber increase. Design cost variations are slight, however. This effect on the demonstrator engine program cost is presented in Fig. 154. The design, fabrication, and test effort does not significantly change in duration due to this variation, so program length is unchanged.

Development Program Schedule and Costs

The development program cost increase with area ratio also is presented in Fig. 154.

First Production Unit Cost

The thrust chamber fabrication cost increase with area ratio results in the first unit cost trend with area ratio presented in Fig. 154.

VARIATION OF THROTTLE CAPABILITY

The baseline engine has a capability of throttling to 20 percent of design thrust. The effects of eliminating the throttle requirement or extending the capability to 10 percent thrust were investigated.

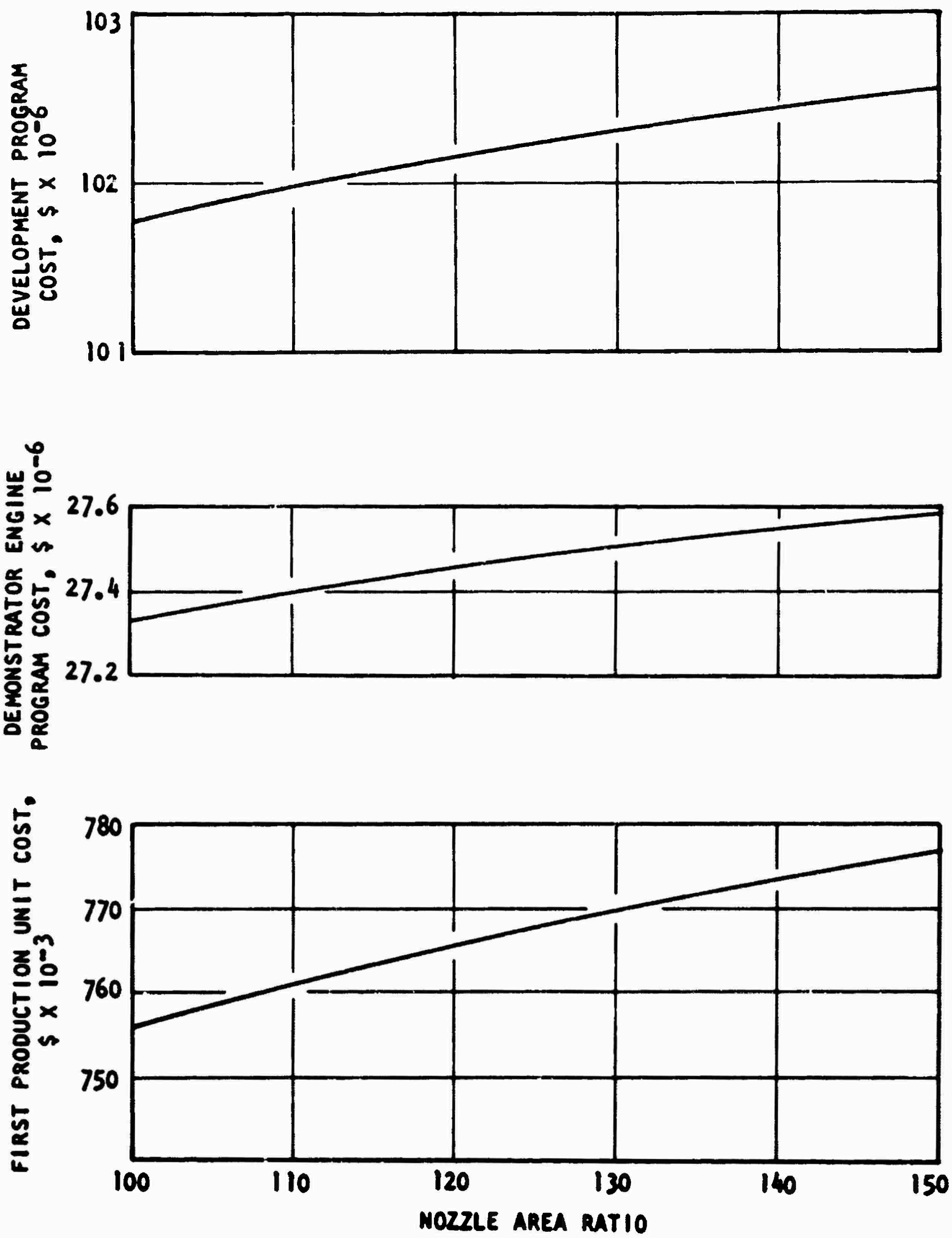


Figure 154. Effect of Single-Panel Nozzle Area Ratio on Costs

Specific Impulse

The variation in design engine performance as a function of the required throttle capability is negligible.

Thrust chamber heat transfer significantly affects performance of the expander turbine drive cycle, but the cooling circuit design is not affected by the required throttle range because the most restrictive operating conditions occur at design thrust (25,000 pounds). Therefore, no changes in the nozzle area ratio, chamber pressure, or cooling circuit are required as the design throttling ratio is varied.

As the engine is throttled, performance decreases as shown in Fig. 155. This curve is applicable for all engines to their lower thrust limit.

Weight

Engine system dry weight is affected slightly by throttling capability, primarily due to design changes to the turbopumps.

If the throttling capability is eliminated, the fuel pump discharge pressure can be reduced approximately 25 psi by reducing the oxidizer turbine control valve pressure drop, resulting in an insignificant weight savings. The fuel pump is the only component whose design is influenced by the 5:1 throttling requirement.

Throttling to 10 percent of design thrust would require redesign of the pumps to prevent operation in the positive-slope region of the head-flow performance maps. By designing the pumps for higher-than-nominal head, operation in the positive-slope region of the head-flow maps can be avoided. The weight penalty is approximately 11 pounds.

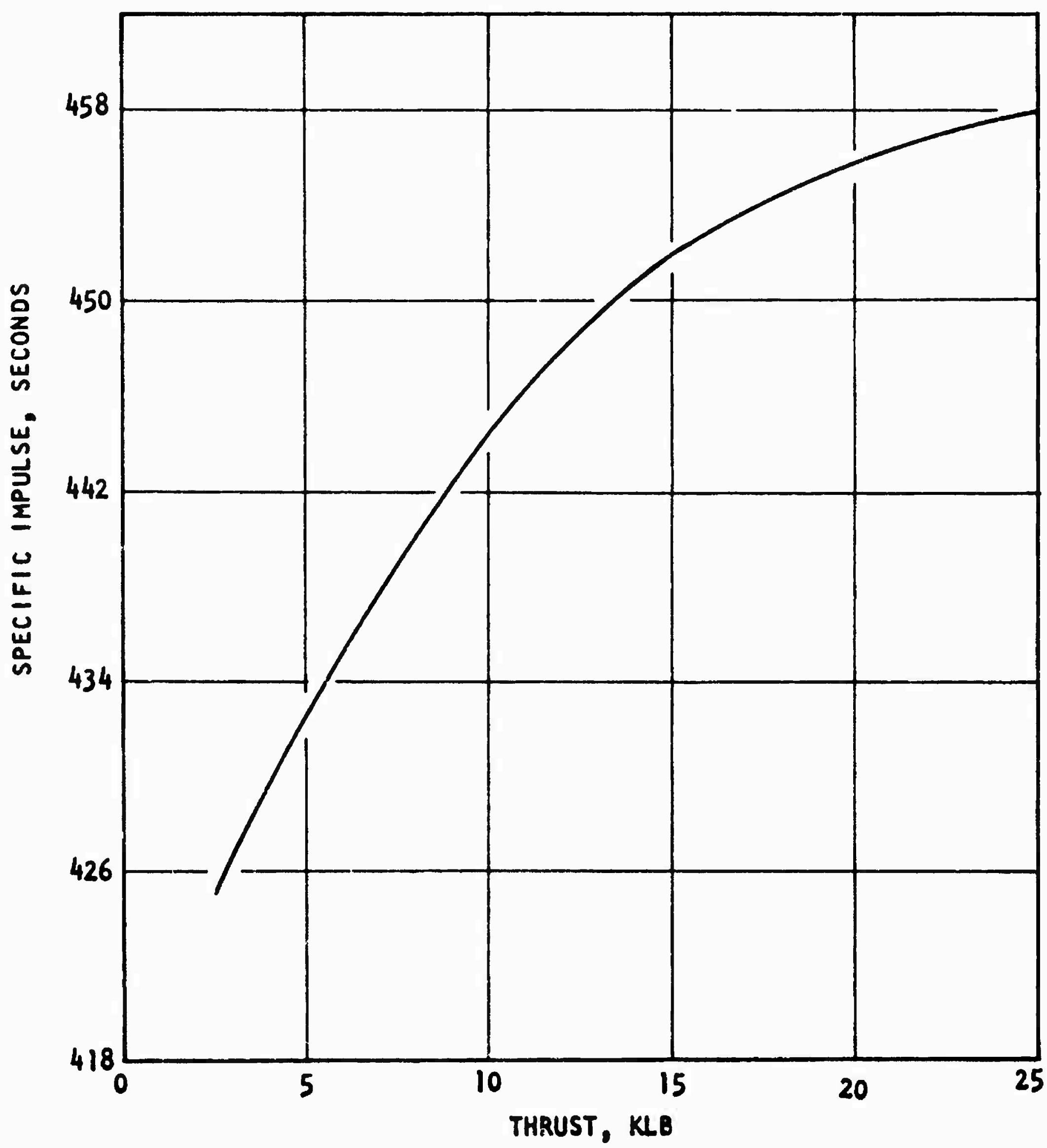


Figure 155. Single-Panel Engine Performance for Throttled Thrust Levels

Cooling Requirements

The most restrictive operating conditions with respect to thrust chamber heat transfer are not affected by throttling requirements.

Start Transient

The start transient is unaffected by elimination of the throttling requirement. The closed-loop control system would be retained to minimize the effects due to variations in the start conditions, i.e., tank pressures and hardware temperatures.

Throttling capability of 10:1 would necessitate a change in the start sequence, but the effects on the transient operating conditions are not significant. At 1.5 and 2.0 seconds into the start transient for the baseline engine, the turbine bypass and oxidizer turbine valves, respectively, are ramped open to predetermined positions to produce a thrust of 5000 pounds, i.e., throttled 5:1. Engine operation is turned over to closed-loop control 1 second later to enable start to the desired thrust level. With increased throttling capability, the only change in the start sequence is the opening of the two control valves at 1.5 and 2.0 seconds to produce 2500 pounds thrust (10:1 throttled) instead of 5000 pounds.

Complexity

Complexity of the engine system is not reduced significantly by eliminating the throttling requirement because the control used for throttling is required for start and mixture ratio control. Consequently, the only change is removal of control system circuitry associated with acceptance of throttling commands by the engine and generation of signals to direct movement of the control valves as a function of desired thrust level. Elimination of the throttling requirement would preclude any concern associated with turbopump critical speed characteristics because the turbopumps would not be required to operate over a range of steady-state speeds.

The operating ranges of the control valves and the injector flow characteristics are within normal design practice to provide sufficient control capability and prevent combustion instability during 10:1 throttling. The increased operating range of the turbopumps will require added attention, however, to investigate critical speed characteristics.

Technology Improvements Needed

No improvements in technology are required for the range of throttling capability considered in this study.

Demonstrator Engine Program Schedule and Costs

Though the effect of eliminating throttle capability on engine design is insignificant, the demonstrator program cost is reduced. This reduction is due to the elimination of throttling-related test activity. The program duration is consequently slightly shorter and the costs associated with test operations, test consumables, and engineering support to test are reduced. Increasing throttling capability from the nominal 5:1 to 10:1 increases these test-related costs because the throttling-related test activity is increased. The somewhat larger turbomachinery needed for the 10:1 throttling also results in added cost. The change in program cost from 5:1 to 10:1 throttling is about 70 percent due to the difference in test requirements, and about 30 percent due to the difference in turbomachinery size-related costs. The effects of throttling on cost are presented in Table 78.

Development Program Schedule and Costs

The engineering development program cost variations with throttling capability also are presented in Table 78. The same factors which cause the demonstrator program cost changes also are responsible for the variations in development program cost.

TABLE 78. EFFECT OF THROTTLING CAPABILITY ON SINGLE-PANEL DEVELOPMENT SCHEDULE AND COSTS

	Throttling Capability		
	None	5:1 (Nom)	10:1
Demonstrator Program Cost, dollars	26.6 M	27.4 M	28.2 M
Δ Cost, dollars	- .8 M	0	+ .8 M
Demonstrator Program Length, months	30.5	31	31.4
Δ Length, months	- .5	0	+ .4
Development Program Cost, dollars	99.0 M	102.0 M	104.7 M
Δ Cost, dollars	-3.0 M	0	+2.7 M
Development Program Length, months	53.2	55	56.2
Δ Length, months	-1.8	0	+1.2
First Unit Cost, dollars	761 K	761 K	775 K
Δ Cost, dollars	0	0	+14 K

First Production Unit Cost

The elimination of throttling capability does not change the first unit cost from its nominal value of $761,000. The larger turbomachinery required for the 10:1 throttling engine increases the cost $14,000 over the nominal 5:1 throttling engine.

IDLE-MODE CAPABILITY

The baseline engine system is not designed for idle-mode operation where both propellants flow directly to the thrust chamber under tank head without rotation of the turbopumps. The design changes required to incorporate this capability into the engine and the performance during idle mode are summarized.

Specific Inpulse

Because the injector and thrust chamber are operating at conditions extremely different from design during tank-head idle mode, an accurate prediction of performance is not appropriate without test data. Delivered specific impulse, based on extrapolations of analytical predictions, is estimated to be approximately 415 seconds.

Weight

If, during engine development, reliable operation of the combustion-wave ignition system cannot be achieved with tank-head propellants, gas storage bottles would be required. The combustion wave igniter storage bottles would be sized for a single start when pressurized to 200 psia; therefore, if idle-mode operation is always followed by a thrust command within the nominal envelope (5000 to 25,000 pounds), no design modifications would be required and recharging of the bottles would proceed as scheduled. If idle mode is to be followed by engine cutoff, the size of the igniter storage bottles must be increased. Sizing the bottles for two starts instead of one would result in a weight penalty of approximately 41 pounds.

Tank-head idle mode requires addition of an on/off, hot-gas valve at the inlet of the fuel turbine to prevent rotation of the fuel turbopump. This valve weighs approximately 10 pounds. If thrust levels which are higher than can be achieved with tank-head propellants are desired, or if the engine mixture ratio is unacceptably low, the driving pressure of the fuel or both propellants must be increased. If this increase is accomplished by allowing rotation of the fuel turbopump, the additional hot-gas valve would not be necessary.

Interface Dimensions

Addition of idle-mode capability does not affect interface dimensions.

Cooling Requirements

Thrust chamber cooling requirements necessitate a reduction in the engine mixture ratio to 3.7 during tank-head idle-mode operation. The resulting coolant bulk temperature is 1460 R at the jacket exit. The heat transfer limits are shown in Fig. 156.

Engine System Operating Conditions

With oxygen and hydrogen tank pressures of 40 and 20 psia, respectively, the nozzle stagnation pressure is approximately 8 psia. Thrust is approximately 225 pounds. If higher thrust levels are desired, or if a mixture ratio of 3.7 is unacceptably low, driving pressure of the fuel or both propellants must be raised to increase the nozzle stagnation pressure. Higher pressures can be achieved by increasing tank pressures or allowing turbopump rotation. Operation of the turbopumps should be avoided, however, because flow oscillations are likely to occur with the pumps throttled to the operating conditions required for idle-mode thrusts.

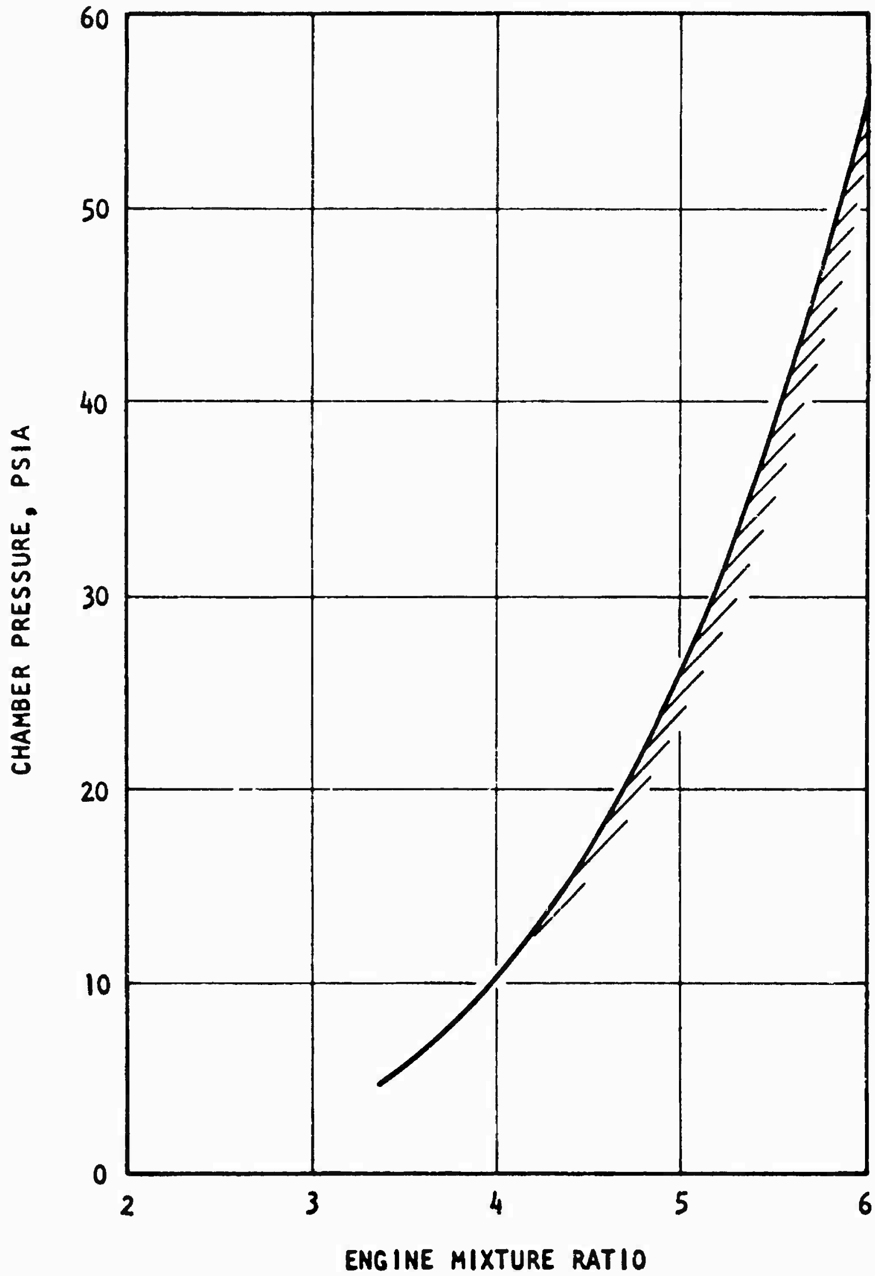


Figure 156. Tank-Head Idle-Mode Heat Transfer Limits for Single-Panel Engine

Start Transient

Tank-head idle-mode capability requires an on-off fuel turbine inlet valve to prevent fuel turbopump rotation and a two-position oxidizer main valve to control mixture ratio. If the main oxidizer valve is not adequate in controlling the small oxidizer flow to the desired accuracy, a small-diameter line and valve in parallel with the main valve will be utilized during idle-mode operation. Sequencing of the control valves must be altered from that of the baseline engine to incorporate idle-mode operation into the start sequence because start to any thrust level from idle-mode operation was assumed.

The modified start sequence is shown in Fig.157. The main fuel and turbine bypass valves are ramped full open to establish fuel coolant flow. Both turbine inlet valves remain closed to prevent turbopump rotation. (In the baseline engine start transient, the turbine bypass valve remains closed for 1.5 seconds and, without a fuel turbine inlet valve, the fuel turbopump is powered by the gaseous coolant flow.) The main oxidizer valve is then opened to its intermediate position and idle-mode operation continues for the desired duration. The engine can then either be shut down or started to a thrust level between 5000 and 25,000 pounds.

Figure 157 also shows a start to full thrust. By opening the fuel turbine inlet valve, and closing the turbine bypass valve until the fuel turbopump is powered to 50 percent of design speed, enough fuel flow is generated to cause the engine to bootstrap to the desired thrust and the main oxidizer valve is then ramped full open. The oxidizer turbine inlet valve is partially opened to allow oxidizer turbopump rotation. Approximately 1 second later, the turbine bypass and oxidizer turbine inlet valves are turned over to closed loop control for the remainder of the start transient. The start time, exclusive of extended idle-mode oepration, is not significantly affected.

Complexity

Complexity of the engine system is not appreciably affected by adding the fuel turbine inlet valve or using a two-position main oxidizer valve. The possibility

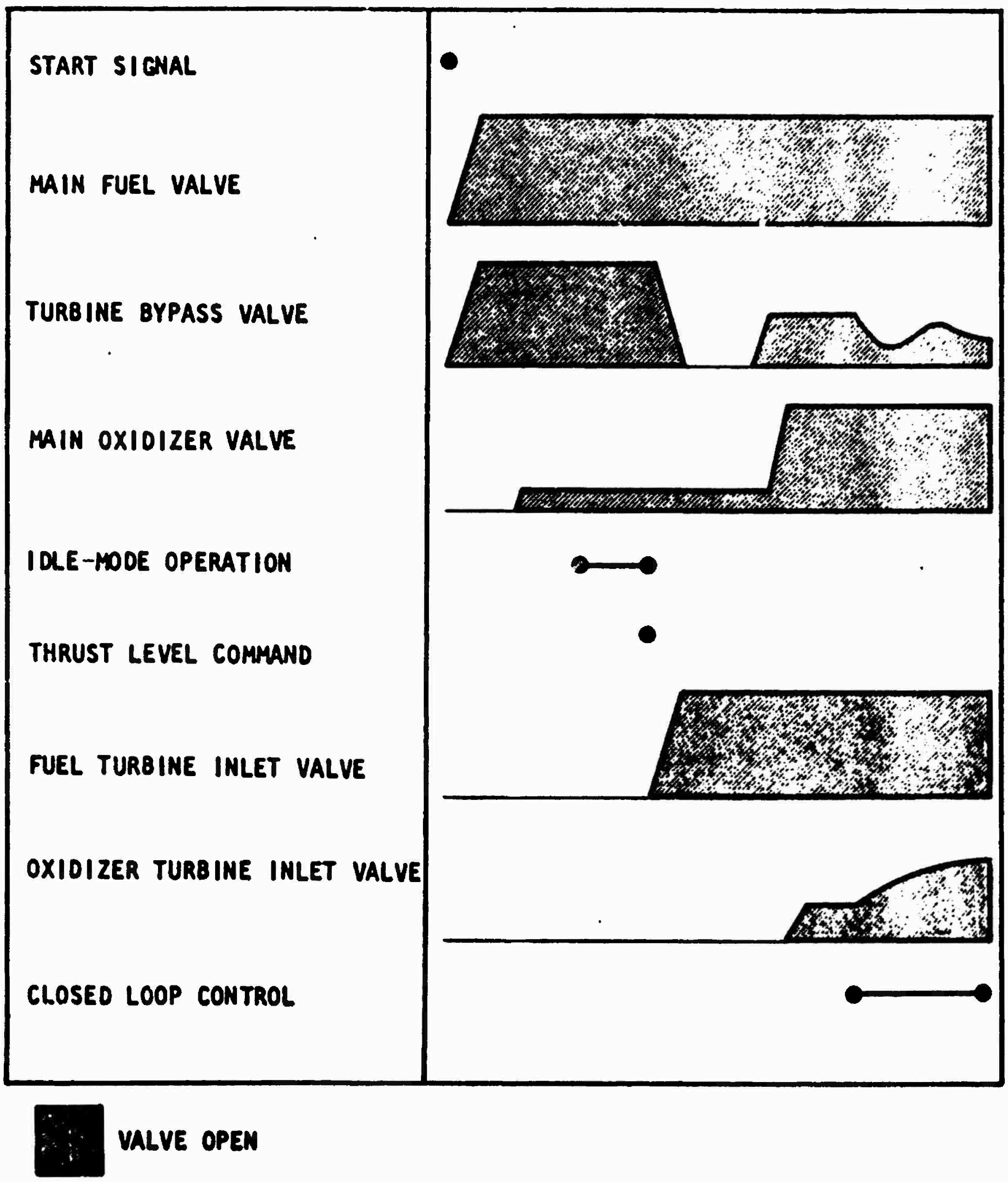


Figure 157. Single-Panel Start Sequence Including Idle-Mode Operation

of combustion instability during pressure-fed operation, however, is a problem which, if preventative corrections are required, could lead to increased complexity of the thrust chamber and injector designs.

Technology Improvements Needed

No improvements in technology are required to provide idle-mode capability. Engine design and development will utilize J-2S idle-mode experience.

Demonstrator Engine Development Schedule and Costs

The engine modifications for idle mode and the effort expended to develop and demonstrate this capability add to the cost of the demonstrator program. The principal cost areas which increase are the valve program because of the new fuel turbine inlet valve and adding two-position control to the main oxidizer valve; the engine test program hardware because the engines now have the added valve costs; and the test operations, test consumables, and engineering test support for the idle-mode-related test activity in the thrust chamber and engine system programs. The cost and program duration effect of idle mode are presented in Table 79.

Development Program Schedule and Costs

Included in Table 79 with the demonstrator cost effect is the development program cost effect. The same areas of cost impact were responsible for the development program cost increase as caused the demonstrator program cost change.

First Unit Cost

The new fuel turbine inlet valve and the addition of two-position capability to the main oxidizer valve adds 30,000 dollars to the cost of the first production unit.

TABLE 79. EFFECT OF IDLE MODE ON SINGLE-PANEL DEVELOPMENT SCHEDULE AND COSTS

	Nominal Engine	Idle Mode Engine
Demonstrator Program Cost, dollars	2.74 M	28.8 M
Δ Cost, dollars	-0-	+ 1.4 M
Demonstrator Program Length, months	31	31.6
Δ Length, months	-0-	+ 0.6
Development Program Cost, dollars	102.0 M	106.7 M
Δ Cost, dollars	-0-	+ 4.7 M
Development Program Length, months	55	57
Δ Length, months	-0-	+ 2
First Unit Cost, dollars	761 K	791 K
Δ Cost, dollars	-0-	+ 30 K

ADVANCED TECHNOLOGY ENGINE CONCEPTS

A review of the 25,000-pound-thrust engine system designs was conducted to determine those technology items that limited the delivered engine performance. Advanced technology concepts (1976-1977 state-of-the-art technology) that would improve the engine design were studied to determine the extent that they would benefit the current engine system design. The most important technology items are those that affect specific impulse and engine weight. Other items might be considered that result in a simplification to the design, fabrication, operation, or reduced maintenance costs. Some of these technology improvements are more near term than others.

The most significant limitation to the aerospike engine performance is the heat transfer limit restricting nozzle expansion area ratio to a maximum value that is a function of thrust level and chamber pressure. Engine size and weight also are a function of chamber pressure and area ratio so to achieve a minimum engine size and weight and high performance, a high area ratio and high chamber pressure are desired. If advanced technology materials and fabrication techniques were made available so that high chamber pressure and high area ratio advanced aerospike engines were feasible, significant improvements could be made in delivered engine performance and engine weight reduction. Improvements in thrust chamber life also could result from this technology advancement.

Increases in turbopump speed would be required for large increases in chamber pressure or pump discharge pressure, and also would result in weight savings by reducing the size of the turbopump. A large-scale design speed increase for the turbopumps could be achievable within the next 5 years with improvements in bearing technology, rotor materials, and fabrication techniques.

A design simplification to the oxidizer pump would be possible for the double-panel engine system by driving the oxidizer turbine with gaseous oxygen from the oxidizer cooling circuit. This drive method would eliminate the critical seal leakage

problem between the pump fluid and the turbine drive gas. The redundant seal package shown in the current oxidizer pump design could be replaced by a floating controlled gap seal, as currently proposed for the hydrogen pump.

ADVANCED TECHNOLOGY MATERIALS

Increasing the chamber pressure and area ratio to 2000 psia and 300:1, respectively, actually decreases the thrust chamber diameter slightly compared to that of the current single-panel engine system because of the compensating effect of chamber pressure and area ratio. A brief engine system design and performance study was conducted to determine the potential performance gains that could be achieved if a refractory material, such as molybdenum-rhenium alloy, were used to fabricate the thrust chamber segments. Coolant channel dimensions of 0.015 in. and a hot-gas wall thickness of 0.015 inch at the throat were required. A maximum operating hot-gas wall temperature of approximately 2300 F resulted. The chamber was considered to be fabricated using the powder metallurgy technique. A delivered engine specific impulse of approximately 485 seconds and a weight of 493 pounds were estimated for this advanced engine system design based on this preliminary investigation.

HIGH-SPEED TURBOMACHINERY

Having selected the chamber pressure and nozzle area ratio for performance and the expander cycle for performance and simplicity, the turbomachinery optimization consists primarily of selecting the speed and number of stages for the components. Adequate efficiencies are obtained with single stages for the turbines and the oxidizer pump. Three stages appear to be a good compromise between the variation of performance and complexity for the fuel pump.

At speeds above approximately 80,000 rpm for the fuel pump, a boost pump will be required to permit efficient operation with an NPSH of 60 feet. Some uncertainty exists in the prediction of efficiencies at the higher speeds, but efficiencies on the order of 70 to 80 percent should be attainable. Pump efficiency does not

influence performance strongly in the closed power cycles, but a higher pump efficiency results in lower turbine power requirements which permit use of turbines with lower pressure ratios which, in turn, result in lower pump discharge pressures and ultimately imply lower turbopump weights. A stronger factor affecting turbopump weights than the pump efficiency is the direct result of increasing the speed of the turbopump. The size and, consequently, the weight of the fuel turbopump decreases markedly with increasing speed, particularly at low speed ranges. At speeds much above 200,000 rpm, the weight decrease is less dramatic. Consequently, 220,000 rpm is seen as an interesting design point for an advanced fuel turbopump. At this speed, the impeller diameter is approximately 2 inches, and the fuel turbopump weighs approximately 35 pounds (compared to approximately 65 pounds at 80,000 rpm). A further reduction of 10 pounds would require increasing the speed to 400,000 rpm.

Technology advances required to permit operation at 220,000 rpm are concerned with the DN value which is a measure of the bearing speed. The upper limit of DN to achieve 10-hour life with rolling contact bearings in LH_2 is approximately 2.0×10^6. Operating at 220,000 rpm with outboard bearings results in a DN value of almost 2.2×10^6. Hydrostatic bearing technology would have to be developed and applied to this operating condition. To gain a performance benefit from hydrostatic bearings in addition to a life benefit, titanium impellers would have to be used because the pump speeds are limited at slightly above the bearing speed limit by the rotor stresses. Titanium impellers are current technology, but the cost is high. Lower cost fabrication techniques could be developed within 5 years.

The efficiency trend curves for the single-stage oxidizer pump are similar to the trends noted for the fuel pump. A low-pressure pump is required for speeds in excess of approximately 25,000 rpm to meet the NPSH requirement of 16 feet. The 10-hour life limit value of DN for the oxidizer pump is approximately 2.0×10^6. To attain a 30-pound reduction in weight, the oxidizer turbopump speed would have to be increased to 160,000 rpm with a DN value of 1.5×10^6. At these conditions, the minimum diameter limit of the impellers of 1.0 inch is met.

Some of this weight saving would be nullified by the boost pump weight. Therefore technology advances are not required to increase the design speed of the current oxidizer turbopump; only the addition of a boost pump to the system would be required.

GASEOUS OXYGEN TURBINE DRIVE

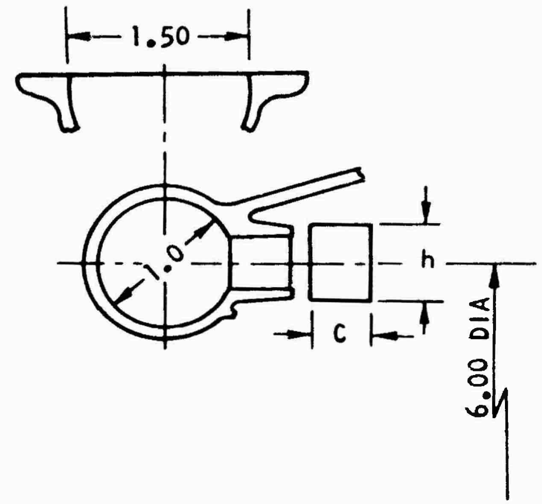
By driving the oxygen turbopump with gaseous oxygen from the double-panel oxidizer cooling circuit, the oxidizer turbopump design could be greatly simplified. The stringent leakage requirement in the current design could be reduced because the same propellant would be present in both the pump and turbine. In the current design, absolute separation of the hydrogen turbine drive gas and the liquid oxygen is achieved with a primary seal, a two-piece purged intermediate seal, and a turbine seal. This seal package can be replaced by a double-floating gap seal similar to that used in the current hydrogen pump design, resulting in a shorter oxygen pump design and additional power margin availability because all of the oxygen is available to drive the oxygen turbopump and all of the hydrogen would be available to drive the hydrogen turbopump. This design modification could be accomplished in a shorter time than the previous items because it could be more properly described as an advanced development effort rather than a technology item.

A brief turbine design study was conducted to determine if the GO_2 drive was practical with the available oxygen flowrate, pressure, and temperature in the current double-panel engine design. Some of the more important design and operating parameters for this turbine are shown in Fig. 158. A GO_2 flowrate of 39.7 lb/sec is required through the turbine, while the total oxidizer flowrate is 45.2 lb/sec. This results in a 5.5 lb/sec, or approximately a 12-percent, GO_2 bypass.

OXIDIZER TURBINE:

GO_2 DRIVE GAS

INLET TEMPERATURE, R	400
INLET PRESSURE, PSIA	1840
EXHAUST TEMPERATURE, R	375
EXHAUST PRESSURE, PSIA	1150
PRESSURE RATIO	1.60
SPEED, RPM	22,000
BLADE SPEED, FT/SEC	577
VELOCITY RATIO	0.782
EFFICIENCY, PERCENT	85
FLOWRATE, LB/SEC	39.7
HORSEPOWER, HP	528
ADMISSION, PERCENT	100
STRESS (N^2AA), X 10^9	8.31



	c	h
NOZZLE	0.350	0.220
ROTOR	0.450	0.911

Figure 158. GO_2 Drive Turbine Preliminary Design and Operating Parameters

DESIGN DATA FOR DIFFERENT THRUST LEVELS

PARAMETRIC ENGINE INFORMATION

Parametric information for O_2/H_2 aerospike engines was generated as a part of this contract effort and has been documented in a separate report, Aerospike Engine Configuration Design and Analysis, Parametric Information. This parametric information report provides information for the following range of aerospike engine design conditions:

Turbine Drive Cycle:	A. Expander Topping
	B. Auxiliary Heat Exchanger
	C. Gas Generator
Engine Mixture Ratio:	5.0, 5.5, 6.0, 7.0:1
Chamber Pressure:	200 to 1000 psia
Nozzle Area Ratio:	50 to 200 (300*)
Thrust Chamber Regenerative cooling concept	Single-Panel and Double-Panel

* Double-Panel Cooling only

The engine system characteristics which were parametrically evaluated are described below. Selected sets of curves from this parametric information report also are presented in this report.

ENGINE CONFIGURATION OPTIMIZATION

This section provides a comparison of the mission performance potential of the different aerospike engine cycles and shows the effect of engine operating parameters (P_c and ε) on mission performance.

DESIGN POINT SPECIFIC IMPULSE

The engine system delivered specific impulse values, which are presented in Fig. 159 through 178 were defined using JANNAF Performance Evaluation Methodology. The specific impulse is for systems operating at their nominal design thrust and mixture ratio.

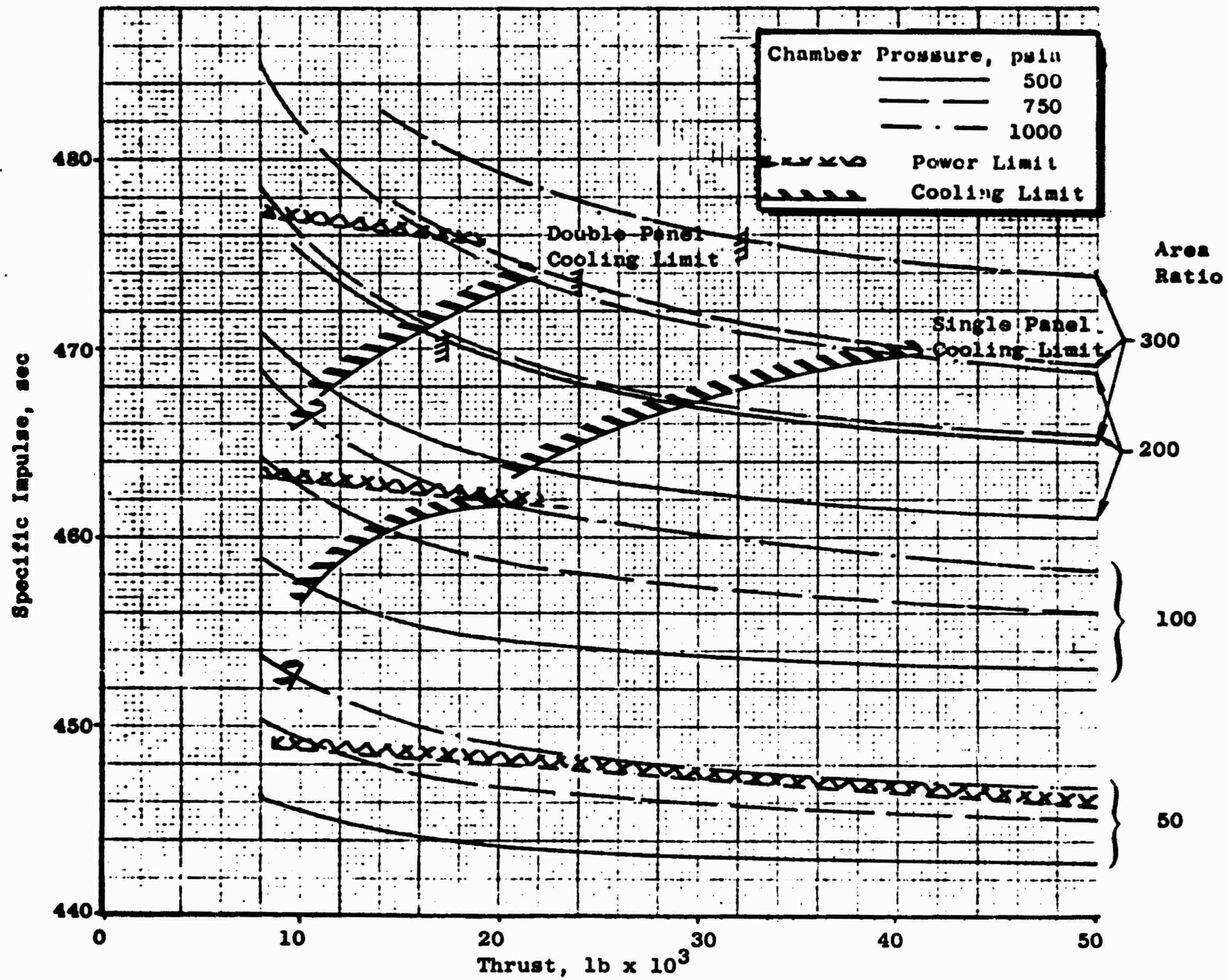


Figure 159. Aerospike Engine Delivered Vacuum Design Performance for Expander Topping Cycle, Double- and Single-Panel Cooling, Engine Mixture Ratio = 5.0

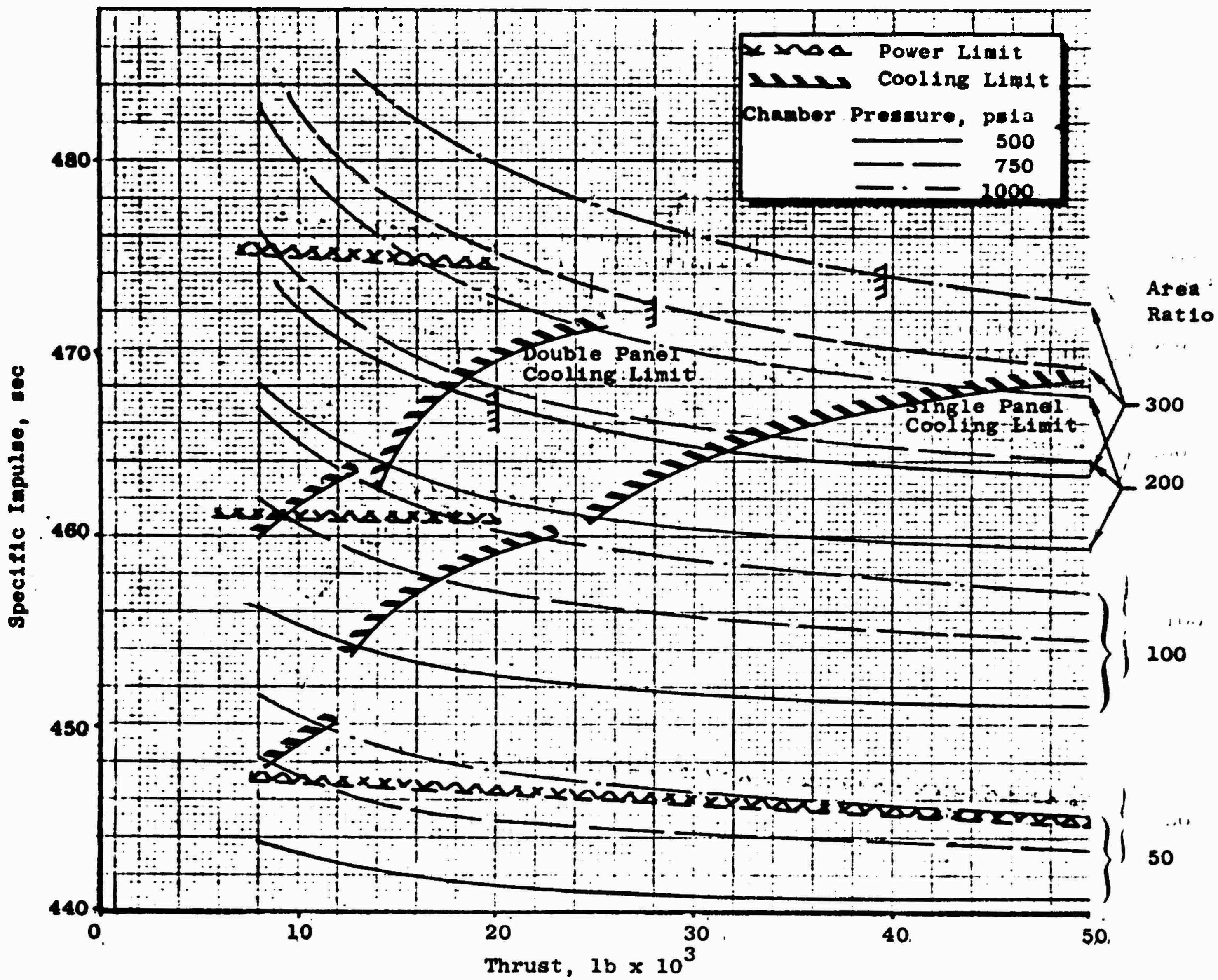


Figure 160. Aerospike Engine Delivered Vacuum Design Performance for Expander Topping Cycle, Double- and Single-Panel Cooling, Engine Mixture Ratio = 5.5

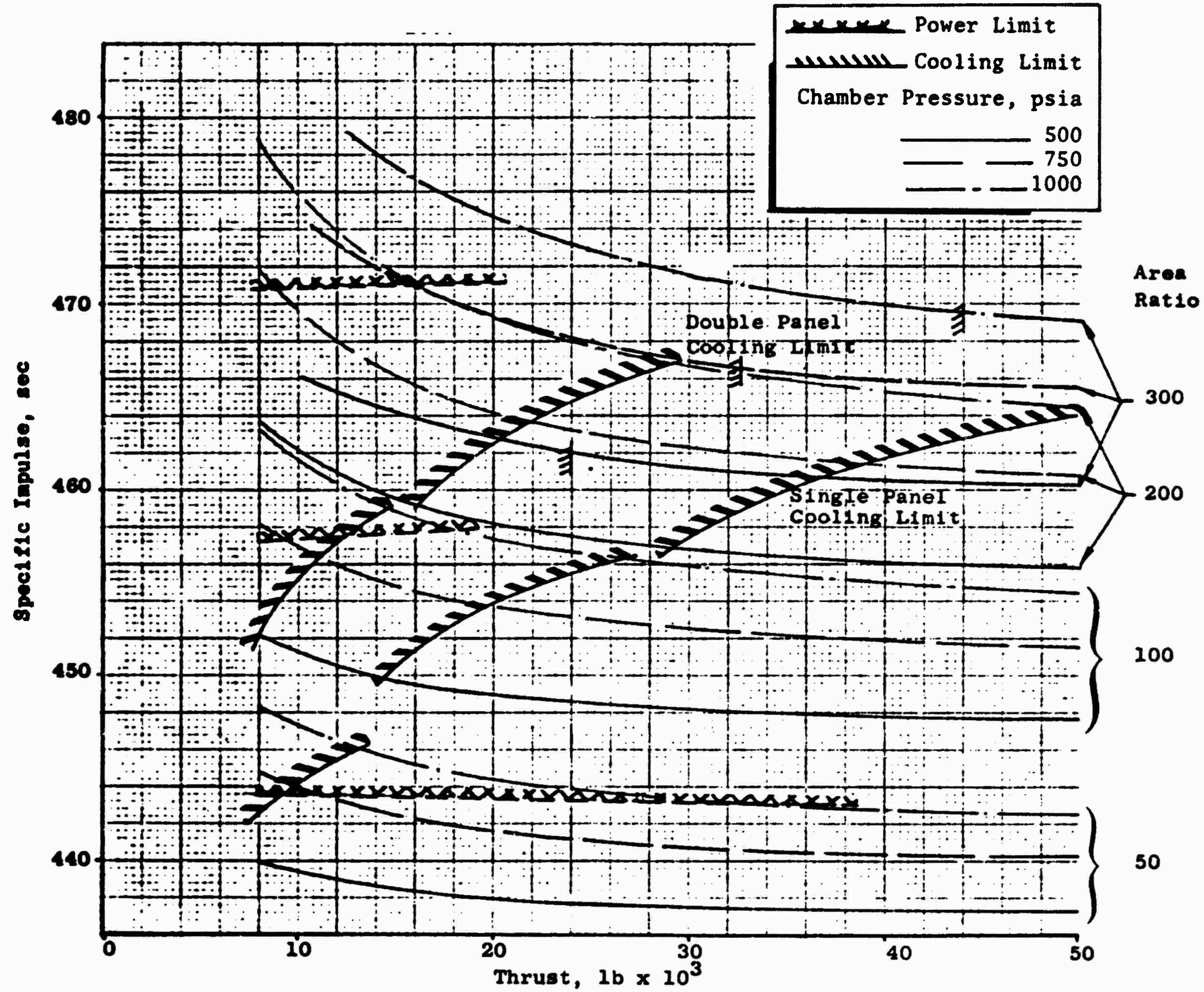


Figure 161. Aerospike Engine Delivered Vacuum Design Performance for Expander Topping Cycle, Double- and Single-Panel Cooling, Engine Mixture Ratio = 6.0

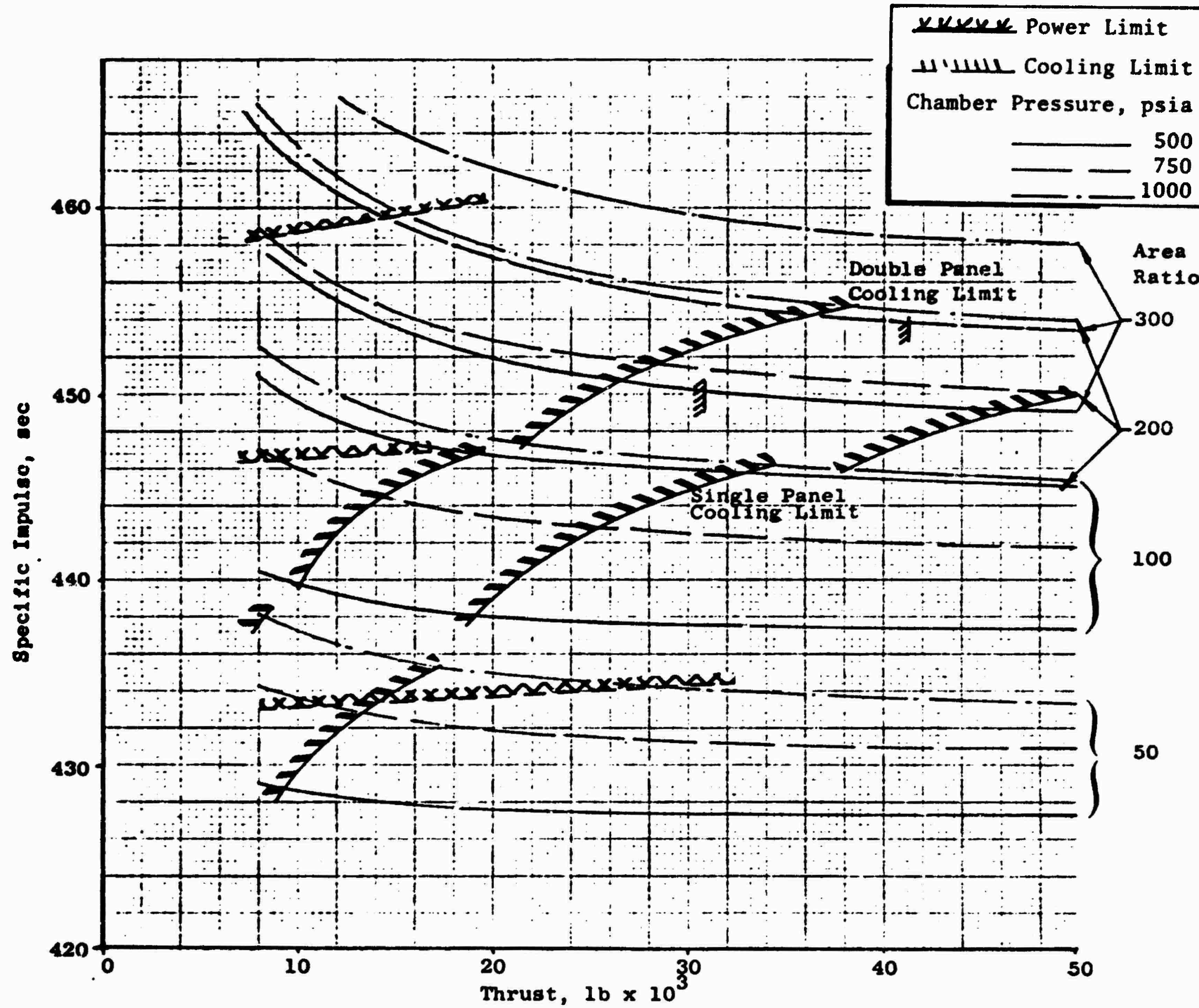


Figure 162. Aerospike Engine Delivered Vacuum Design Performance for Expander Topping Cycle, Double- and Single-Panel Cooling, Engine Mixture Ratio = 7.0

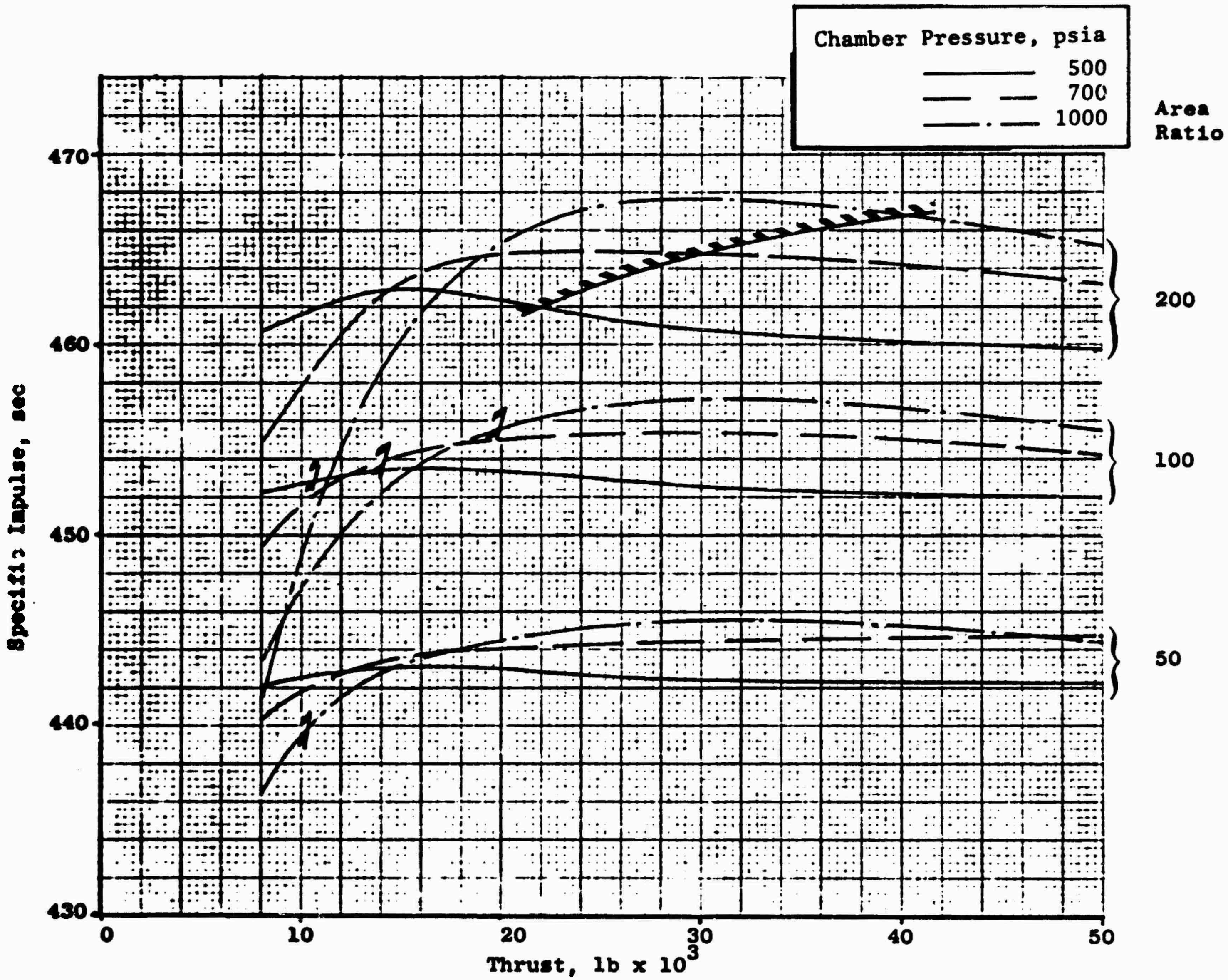


Figure 163. Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Single-Panel Cooling, Engine Mixture Ratio = 5.0

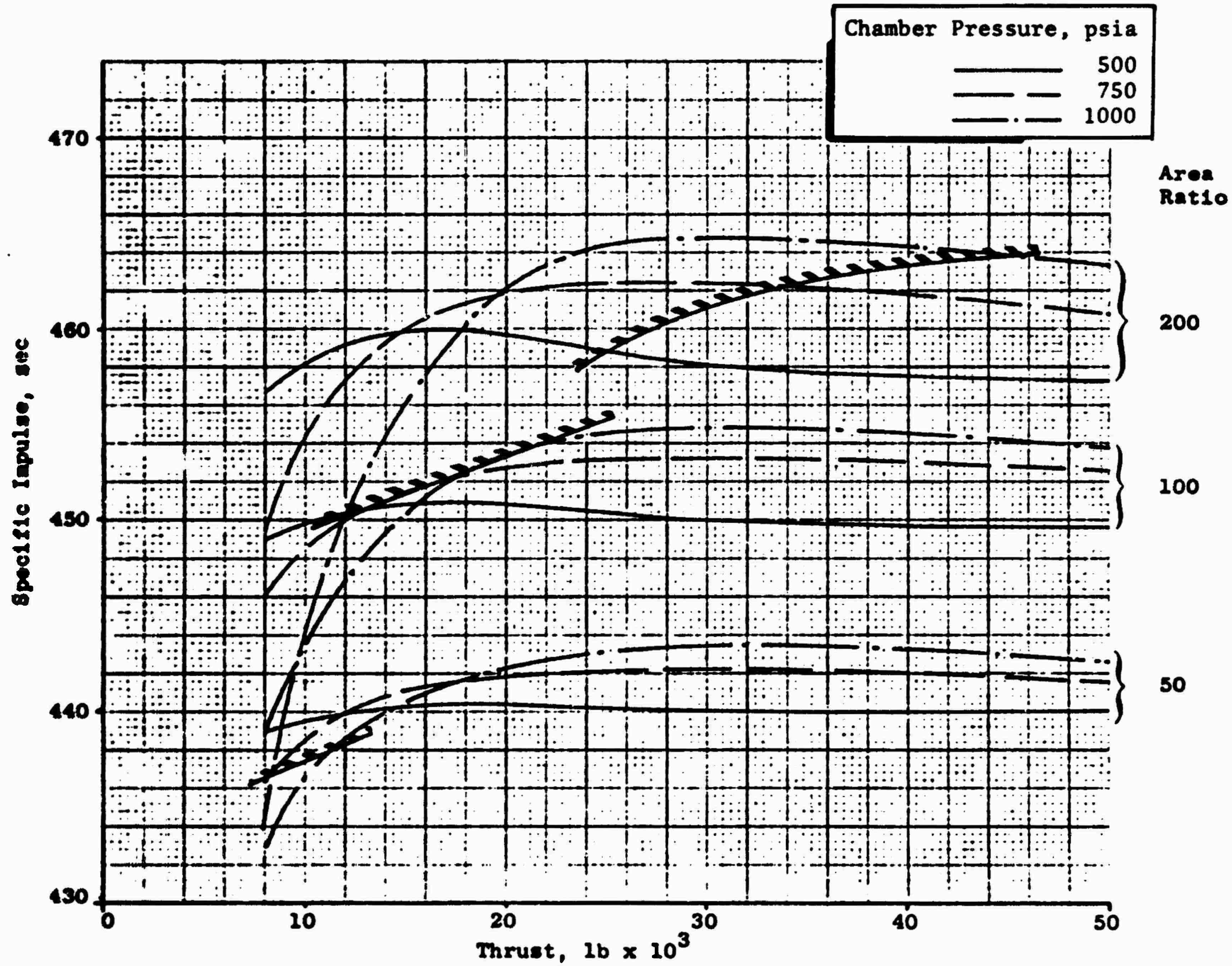


Figure 164. Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Single-Panel Cooling, Engine Mixture Ratio = 5.5

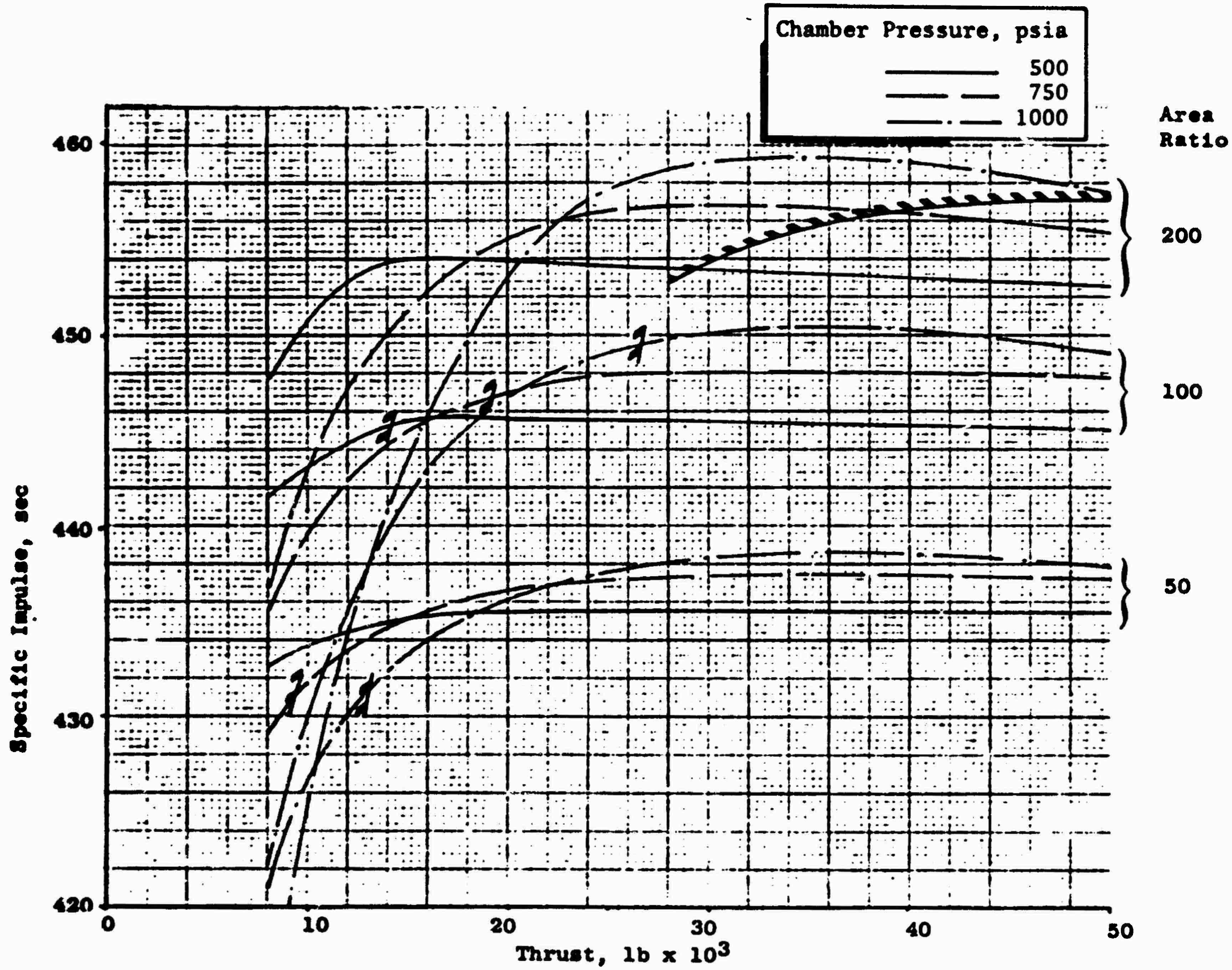


Figure 165. **Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Single-Panel Cooling, Engine Mixture Ratio = 6.0**

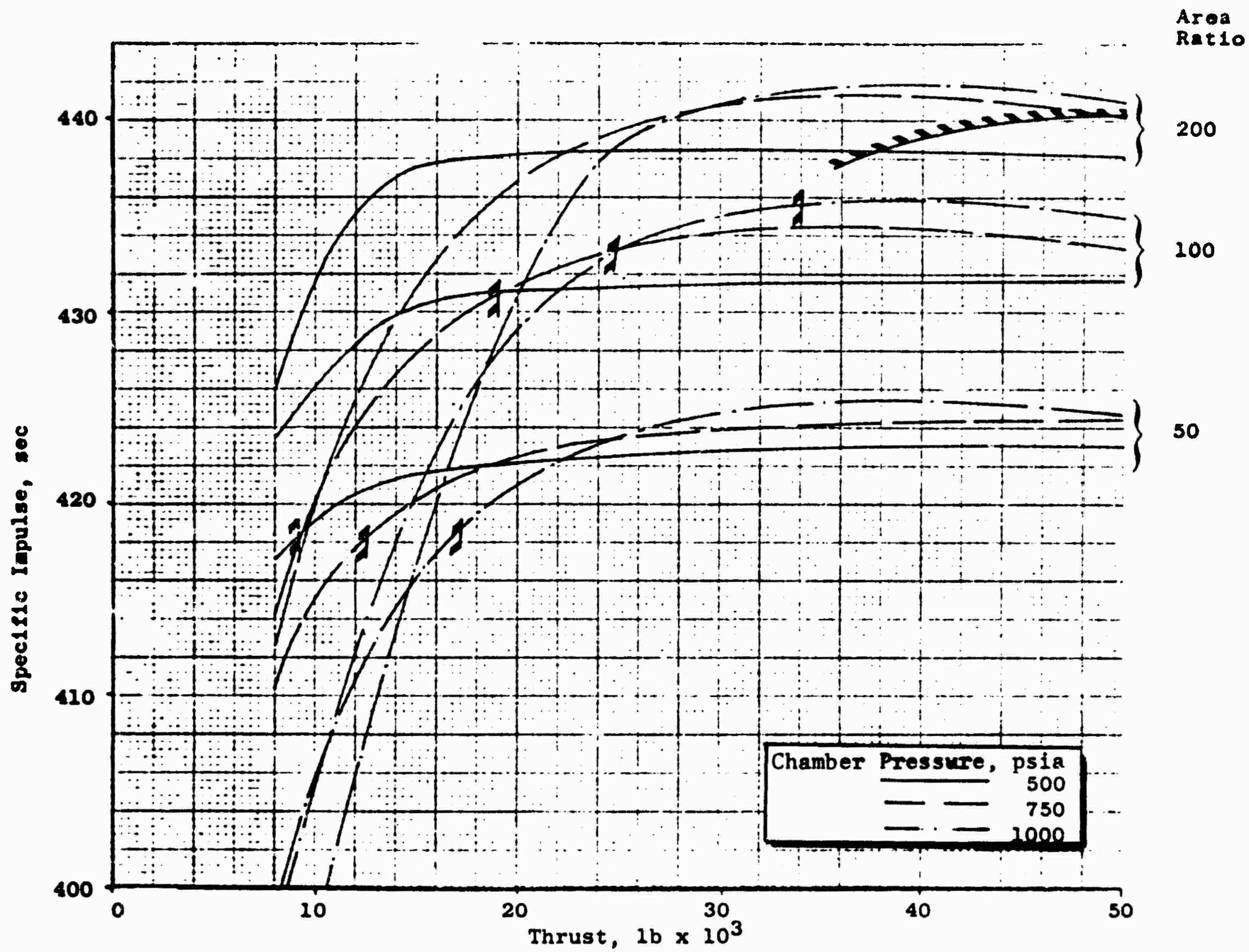


Figure 166. Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Single-Panel Cooling, Engine Mixture Ratio = 7.0

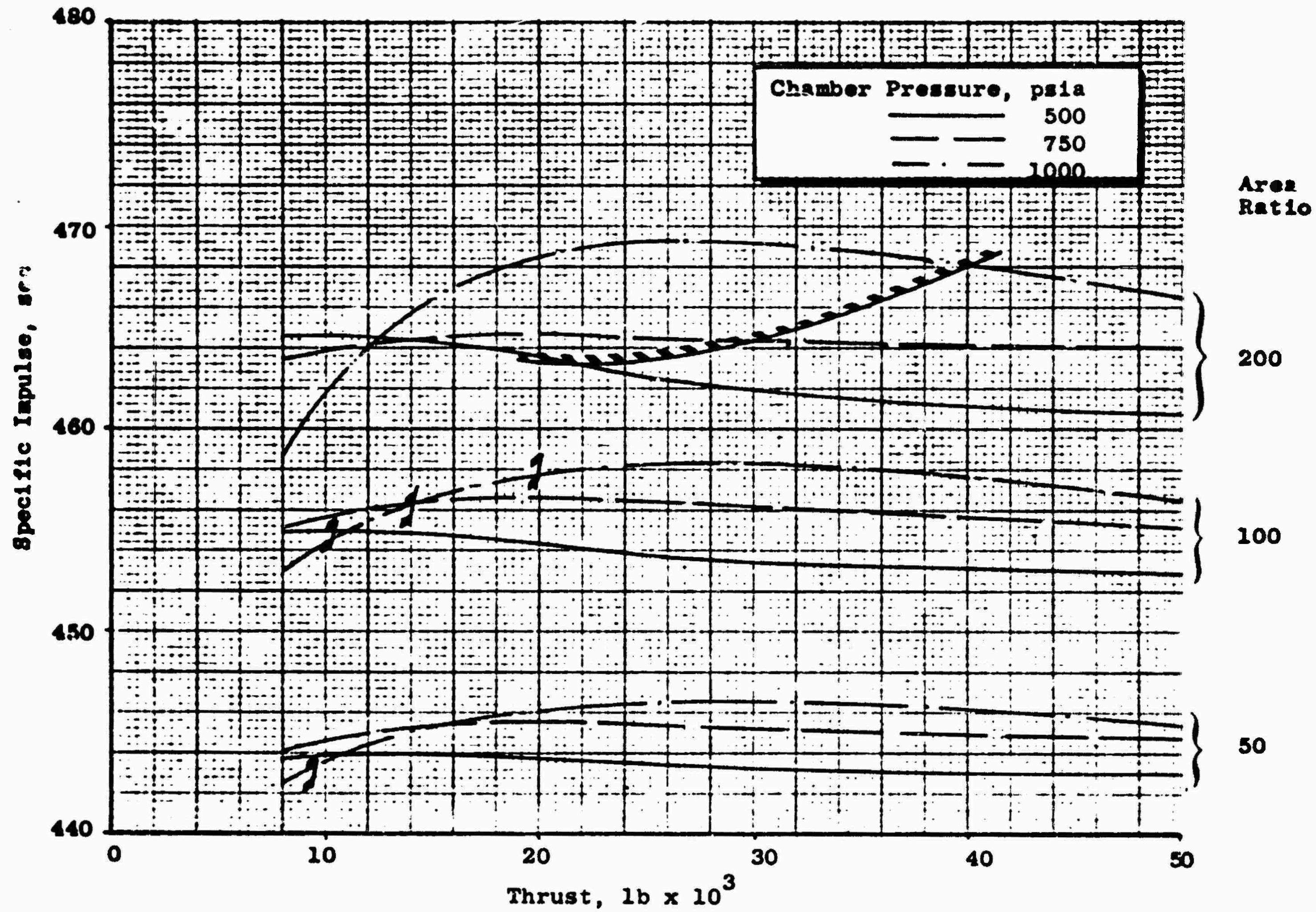


Figure 167. Aerospike Engine Delivered Vacuum Design Performance for Gas Gene-ator Cycle, Single-Panel Cooling, Engine Mixture Ratio = 5.0

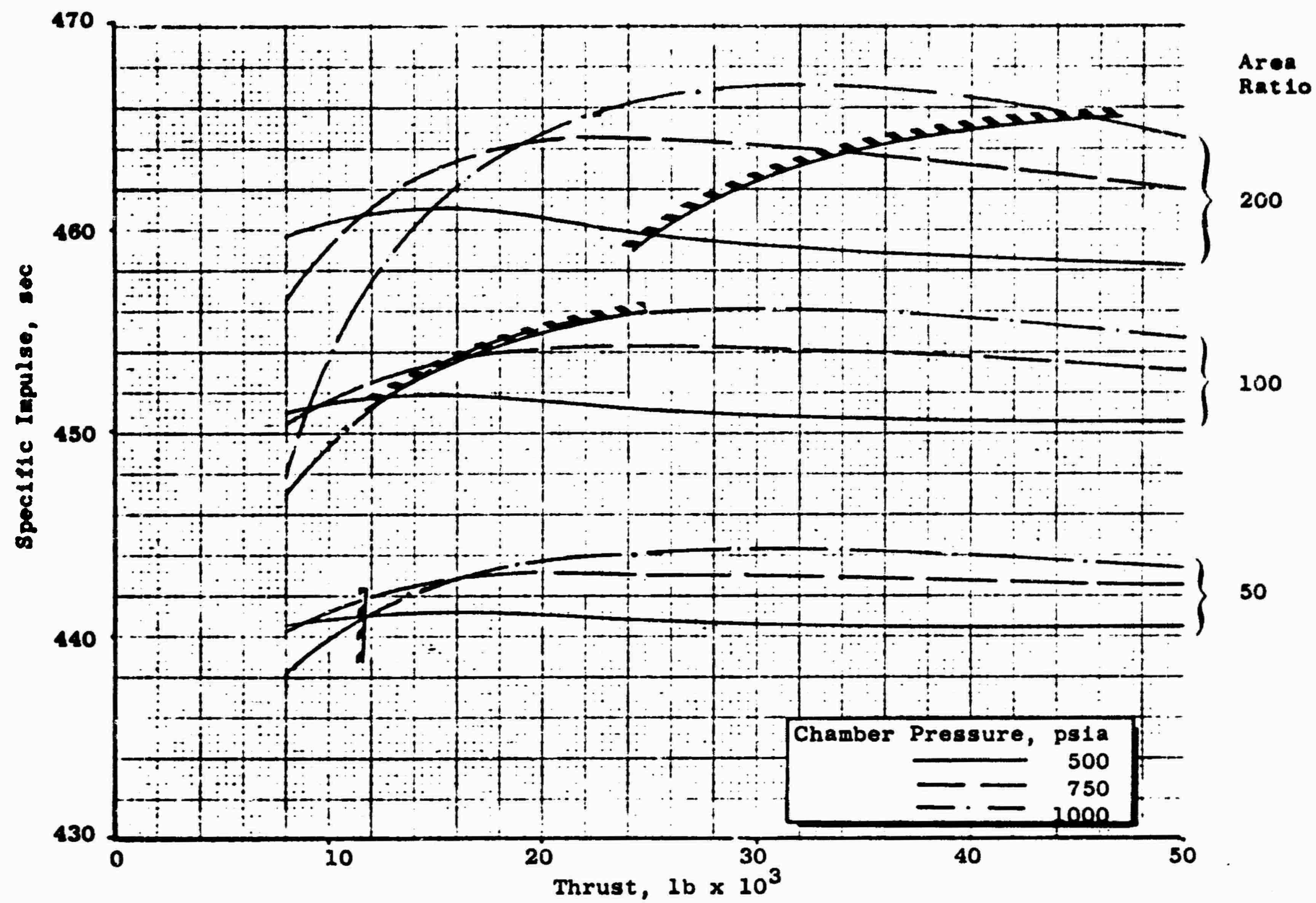


Figure 168. **Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Single-Panel Cooling, Engine Mixture Ratio = 5.5**

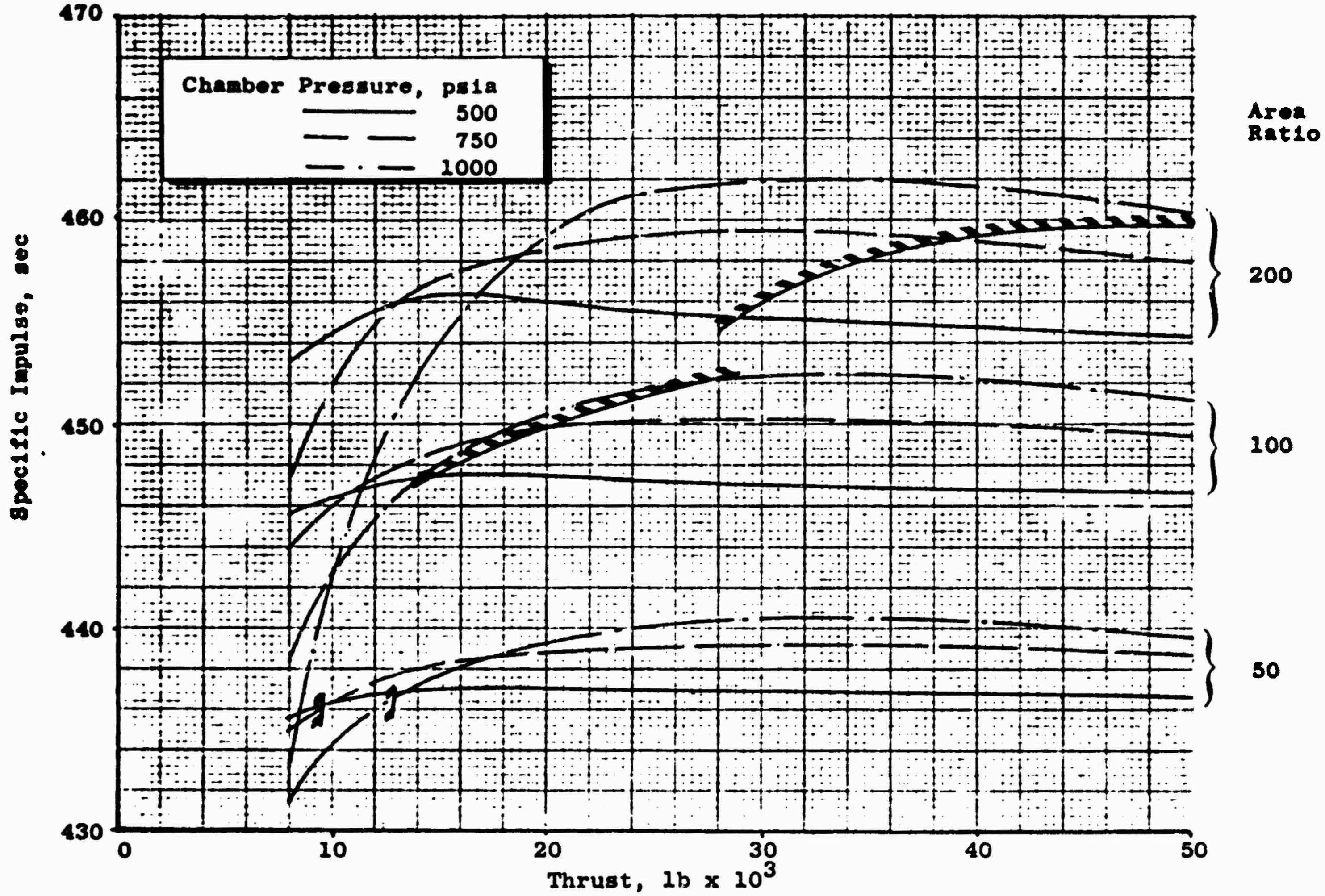


Figure 169. Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Single-Panel Cooling, Engine Mixture Ratio = 6.0

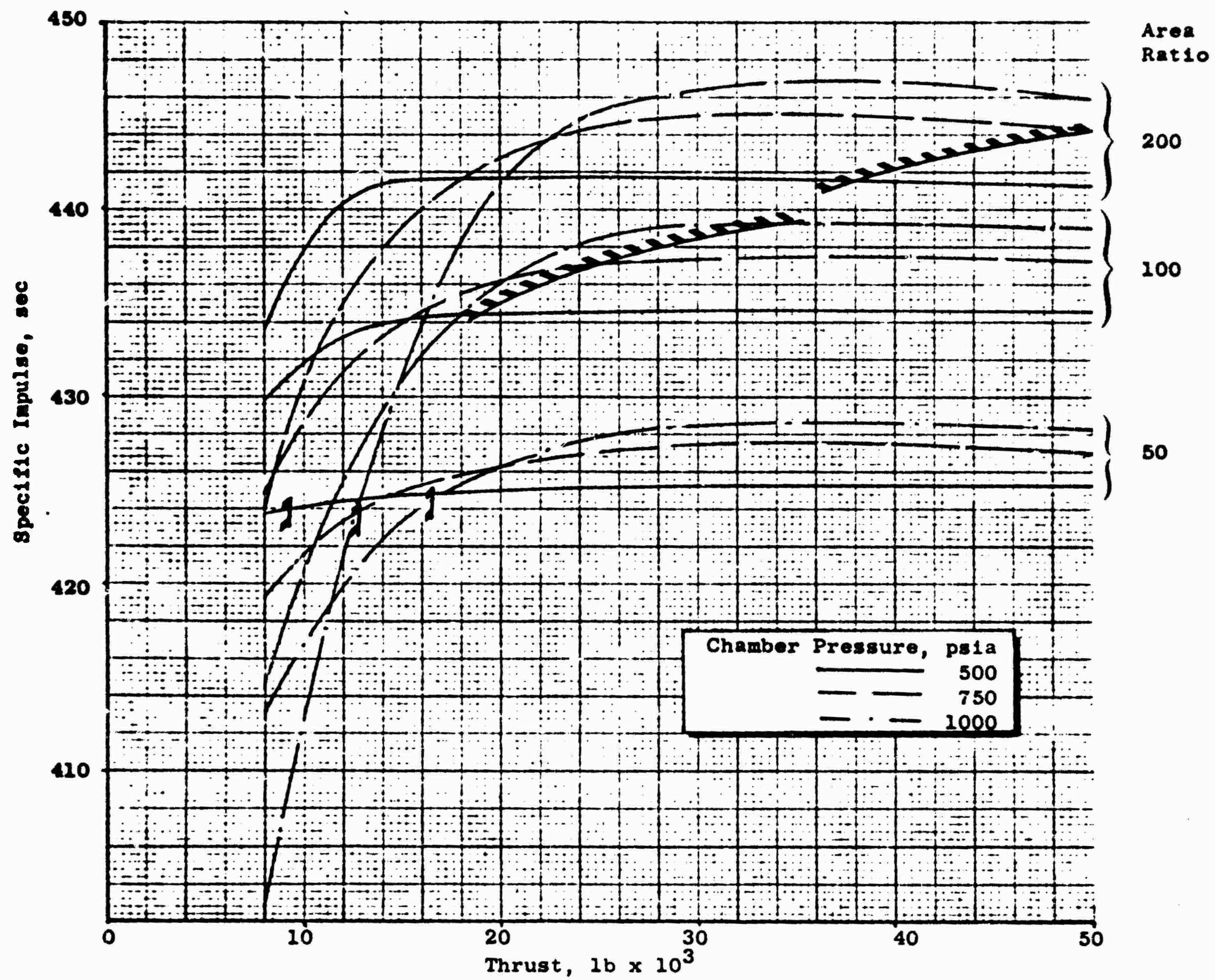


Figure 170. Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Single-Panel Cooling, Engine Mixture Ratio = 7.0

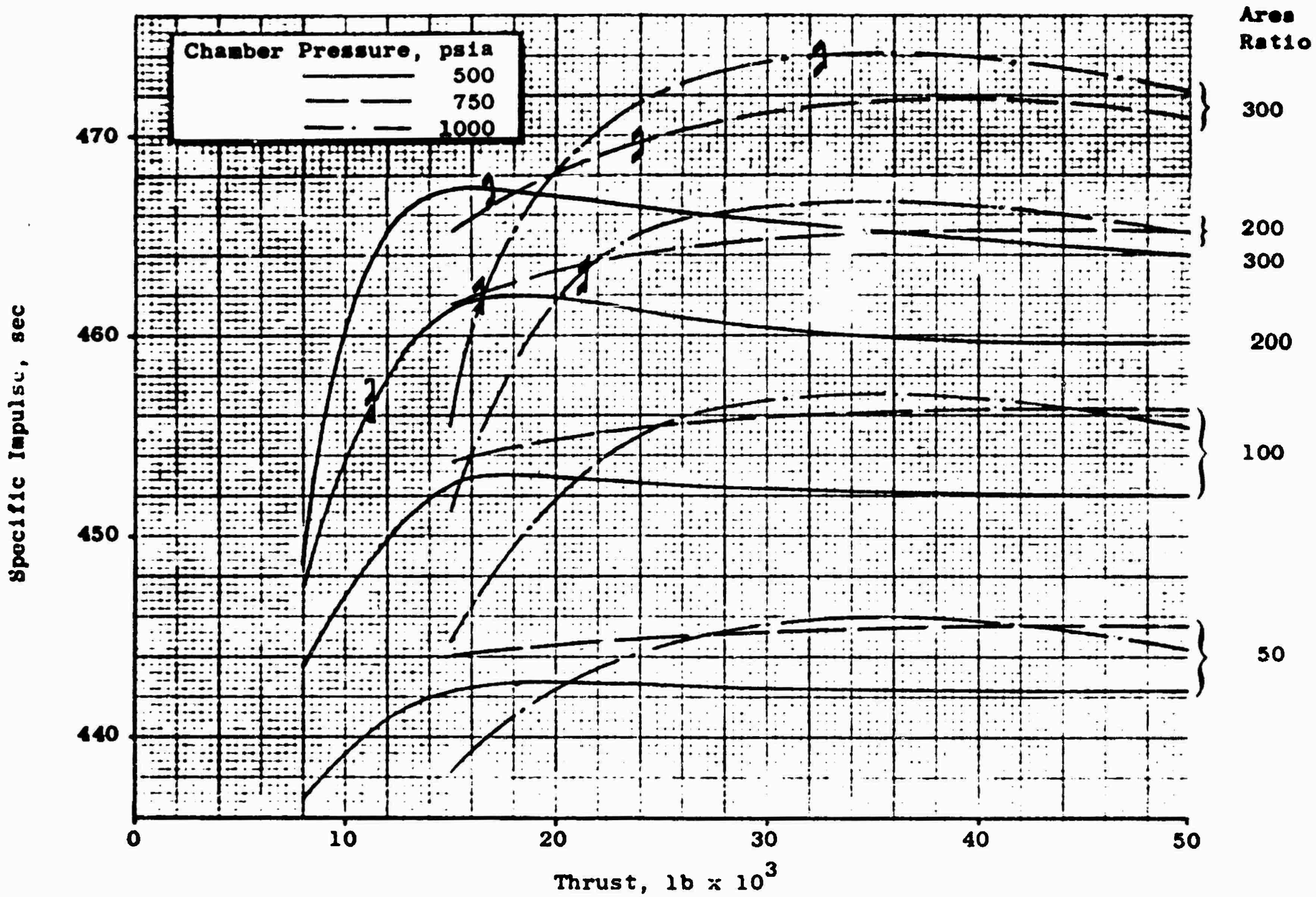


Figure 171. Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Double-Panel Cooling, Engine Mixture Ratio = 5.0

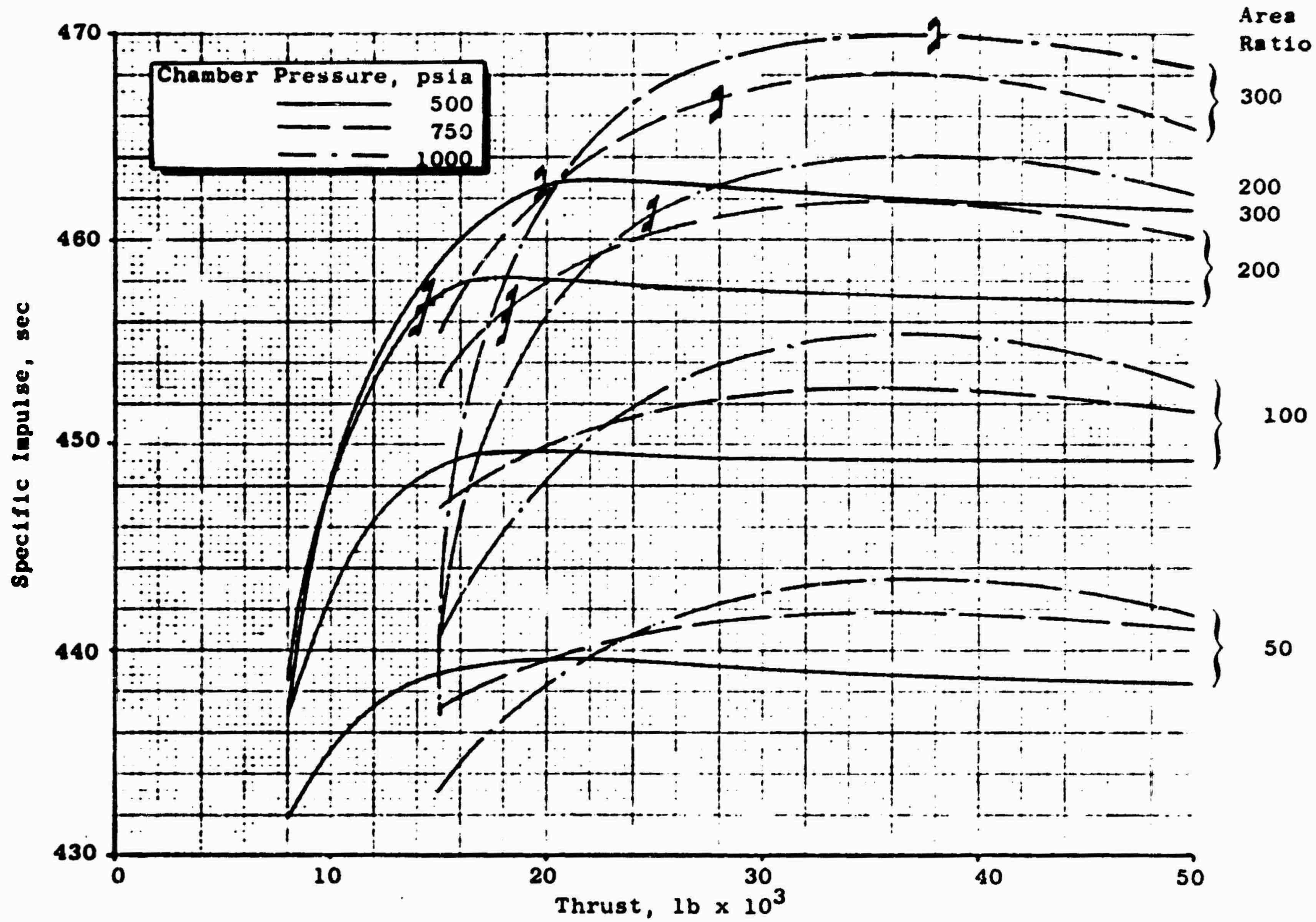


Figure 172. **Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Double-Panel Cooling, Engine Mixture Ratio = 5.5**

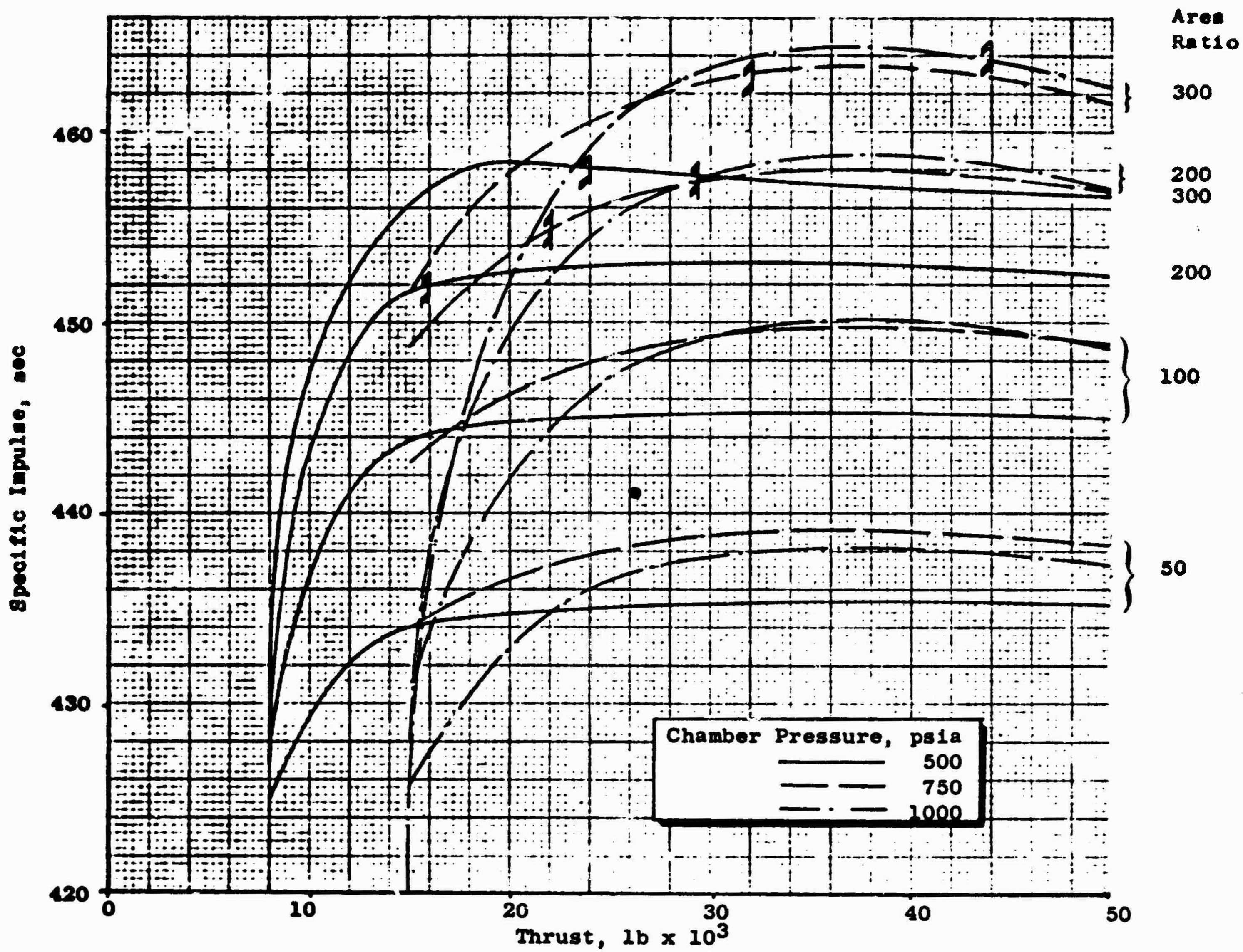


Figure 173. Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Double-Panel Cooling, Engine Mixture Ratio = 6.0

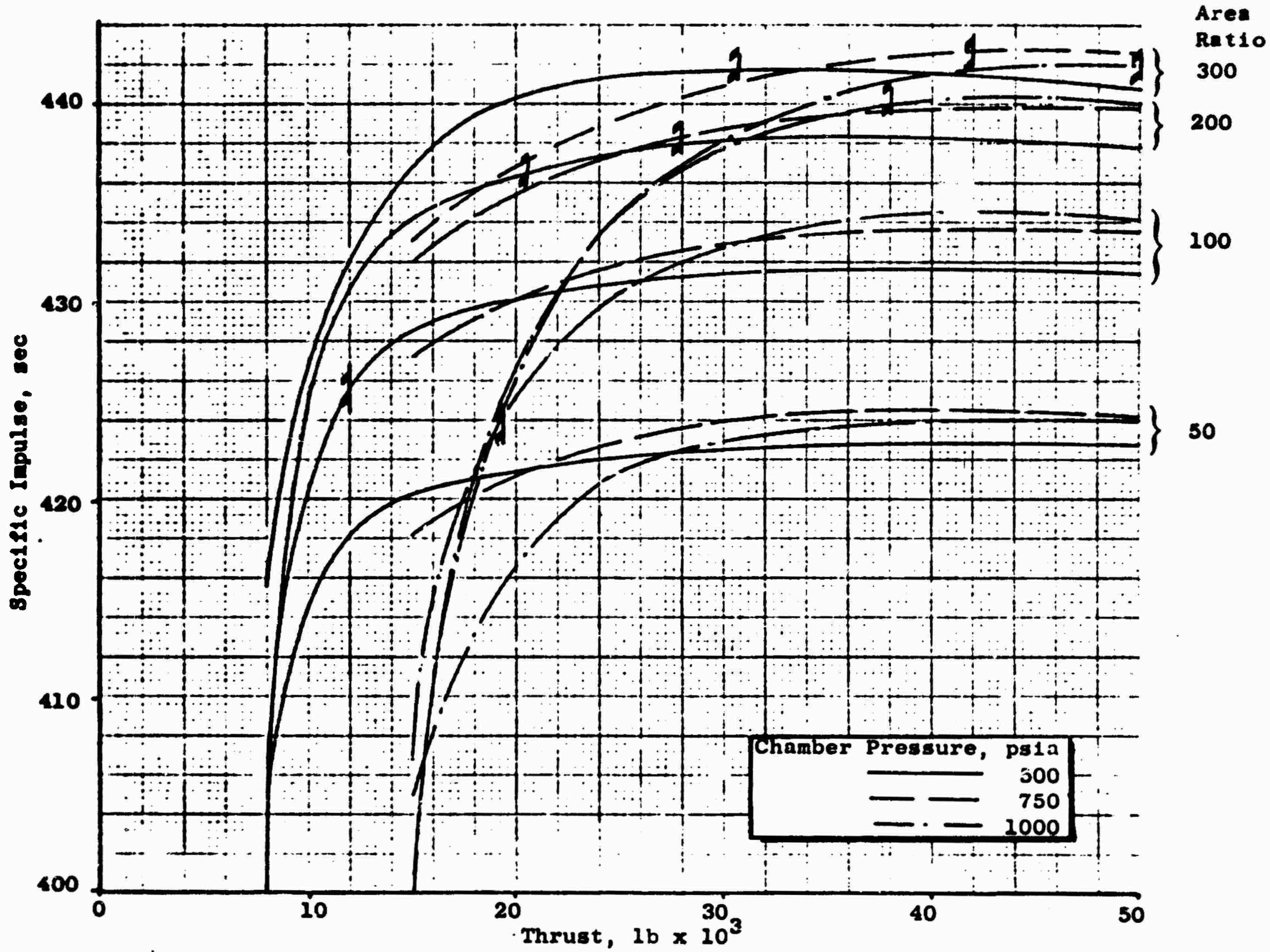


Figure 174. Aerospike Engine Delivered Vacuum Design Performance for Auxiliary Heat Exchanger Cycle, Double-Panel Cooling, Engine Mixture Ratio = 7.0

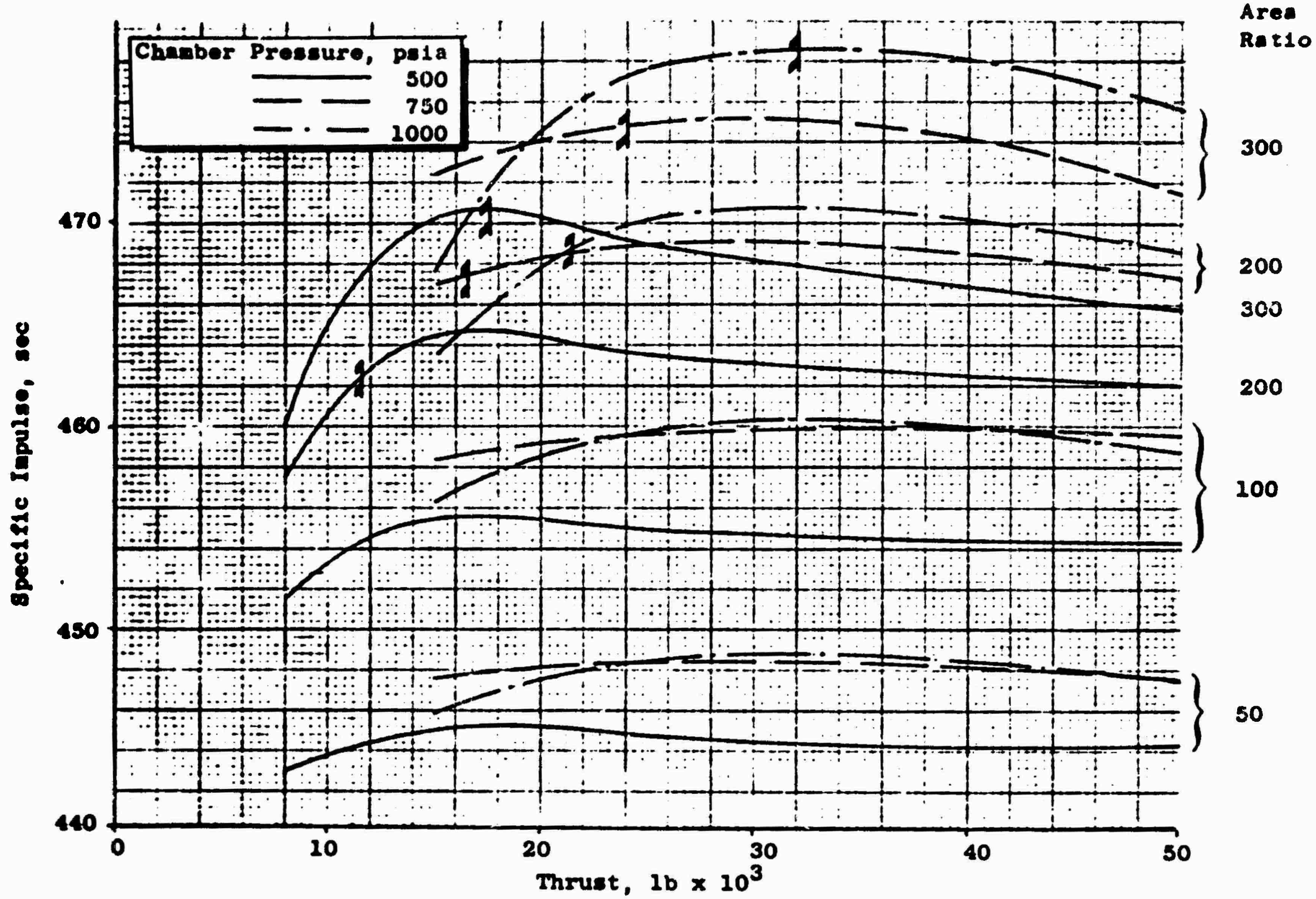


Figure 175. Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Double-Panel Cooling, Engine Mixture Ratio = 5.0

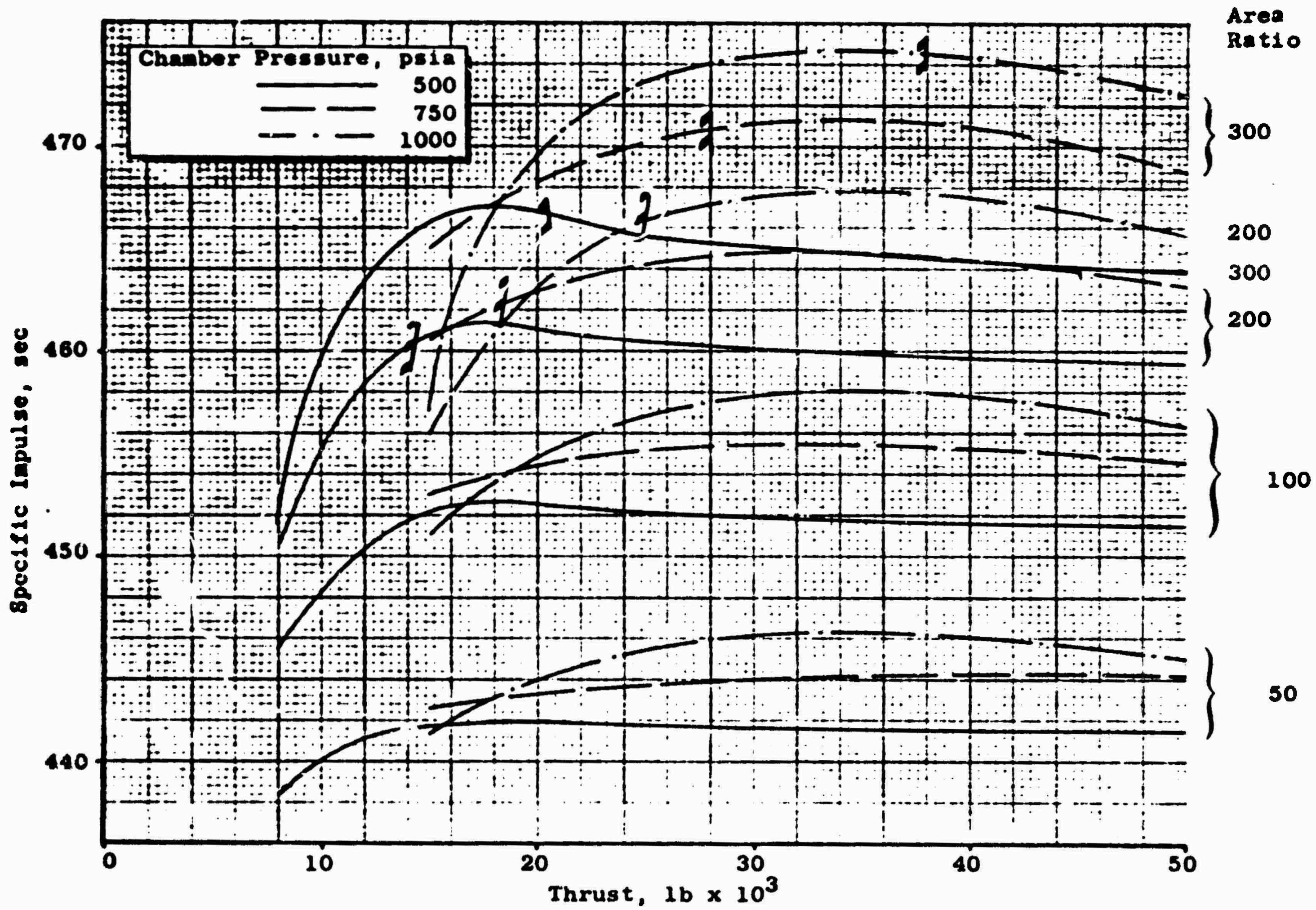


Figure 176. Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Double-Panel Cooling, Engine Mixture Ratio = 5.5

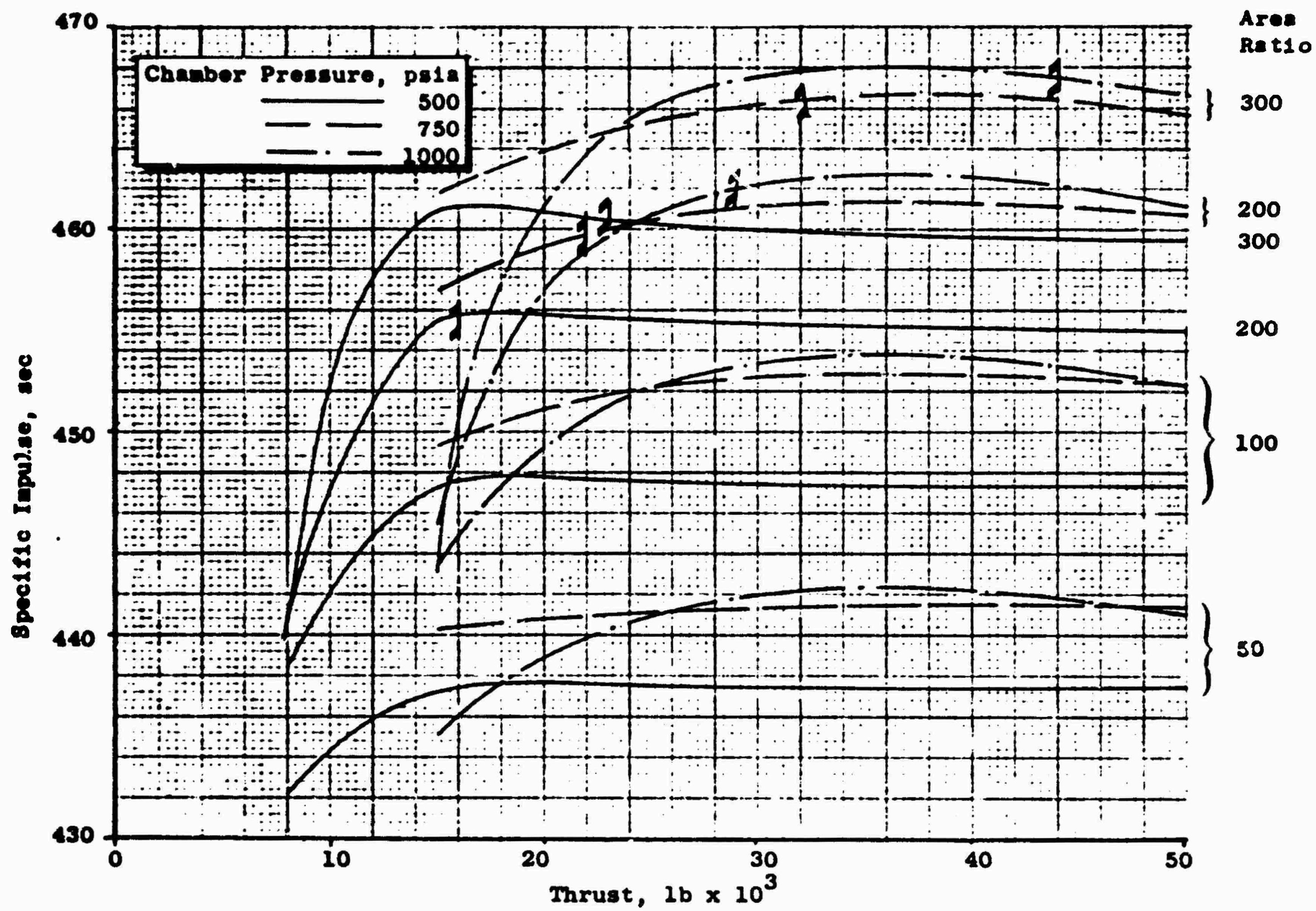


Figure 177. **Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Double-Panel Cooling, Engine Mixture Ratio = 6.0**

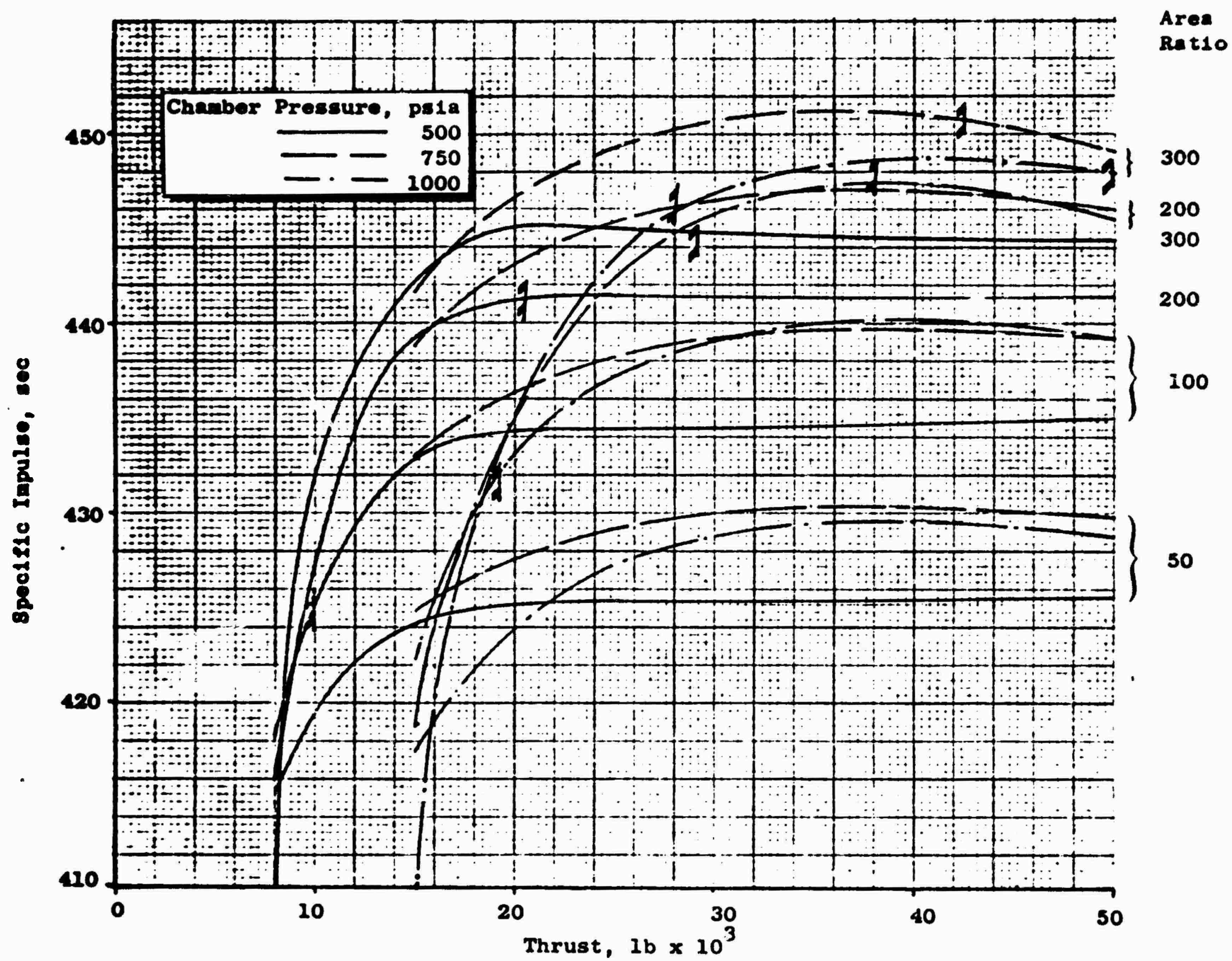


Figure 178. **Aerospike Engine Delivered Vacuum Design Performance for Gas Generator Cycle, Double-Panel Cooling, Engine Mixture Ratio = 7.0**

DESIGN POINT ENGINE WEIGHT

The total dry engine system weights are presented in Fig. 179 through 190 for the aerospike engine configurations.

ENGINE SYSTEM DIMENSIONS

Engine length (gimbal to nozzle exit) and overall diameter are presented in Fig. 191 through 198.

OFF-DESIGN SPECIFIC IMPULSE

The aerospike engine system configurations are capable of operating over a ±0.5 mixture ratio range and have a throttling capability of 5:1. The delivered specific impulse variations over this operating envelope are defined.

ENGINE NPSH EFFECTS

This section shows the effect of variations in the nominal design value of engine mixture ratio and available NPSH or engine performance, weight, dimensions, and turbopump operating characteristics.

THERMAL CONDITIONING REQUIREMENTS

The total propellant flow required to chill the engine inlet ducting and pump for an acceptable start are presented for a range of thrust levels and chamber pressures.

ENGINE COST

The cost of engine development, including a certification program, and the cost of the first engine production unit are defined in this section.

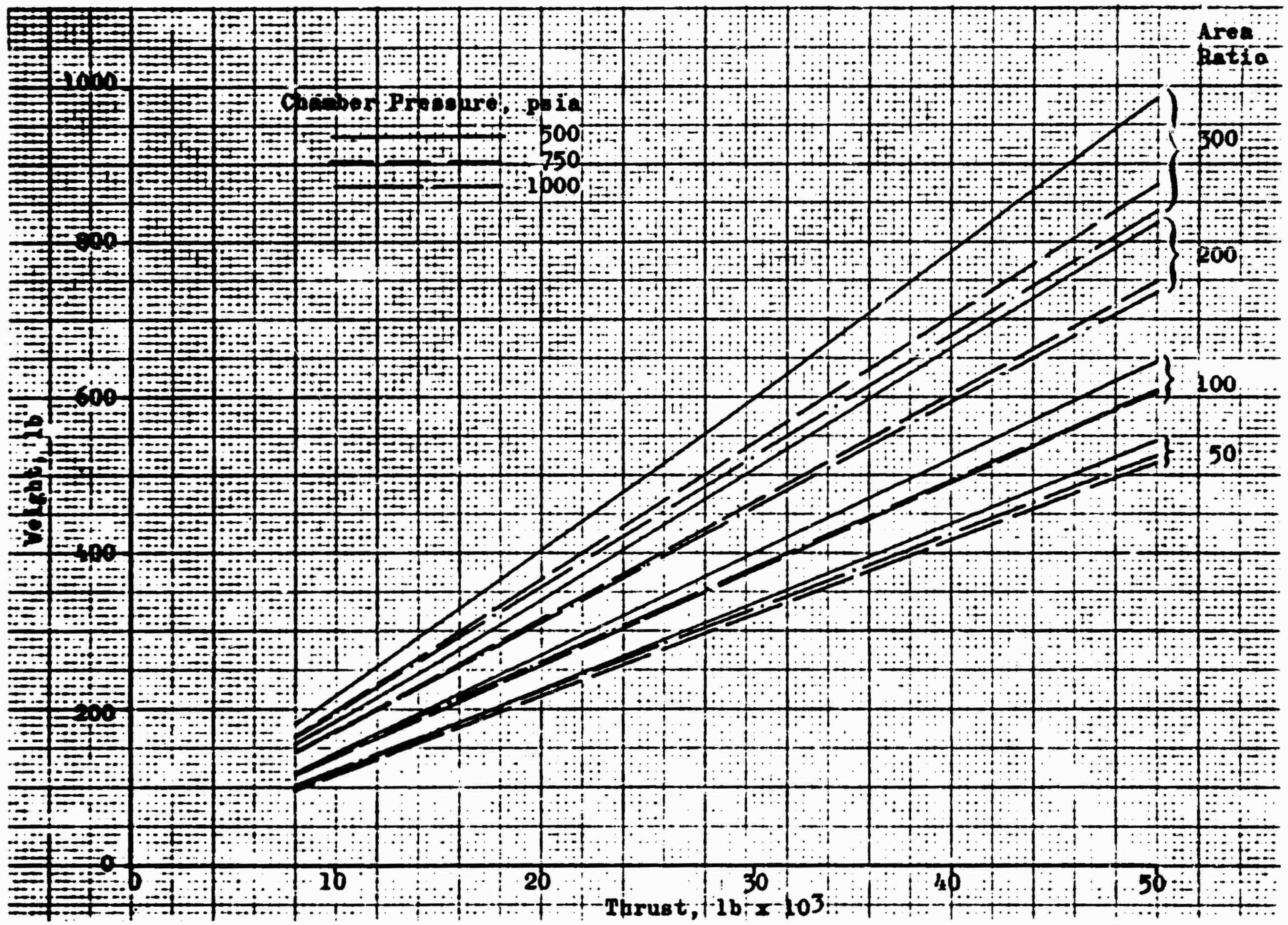


Figure 179. Aerospike Engine System Dry Weight for Expander Togging Cycle, Engine Mixture Ratio = 5.0

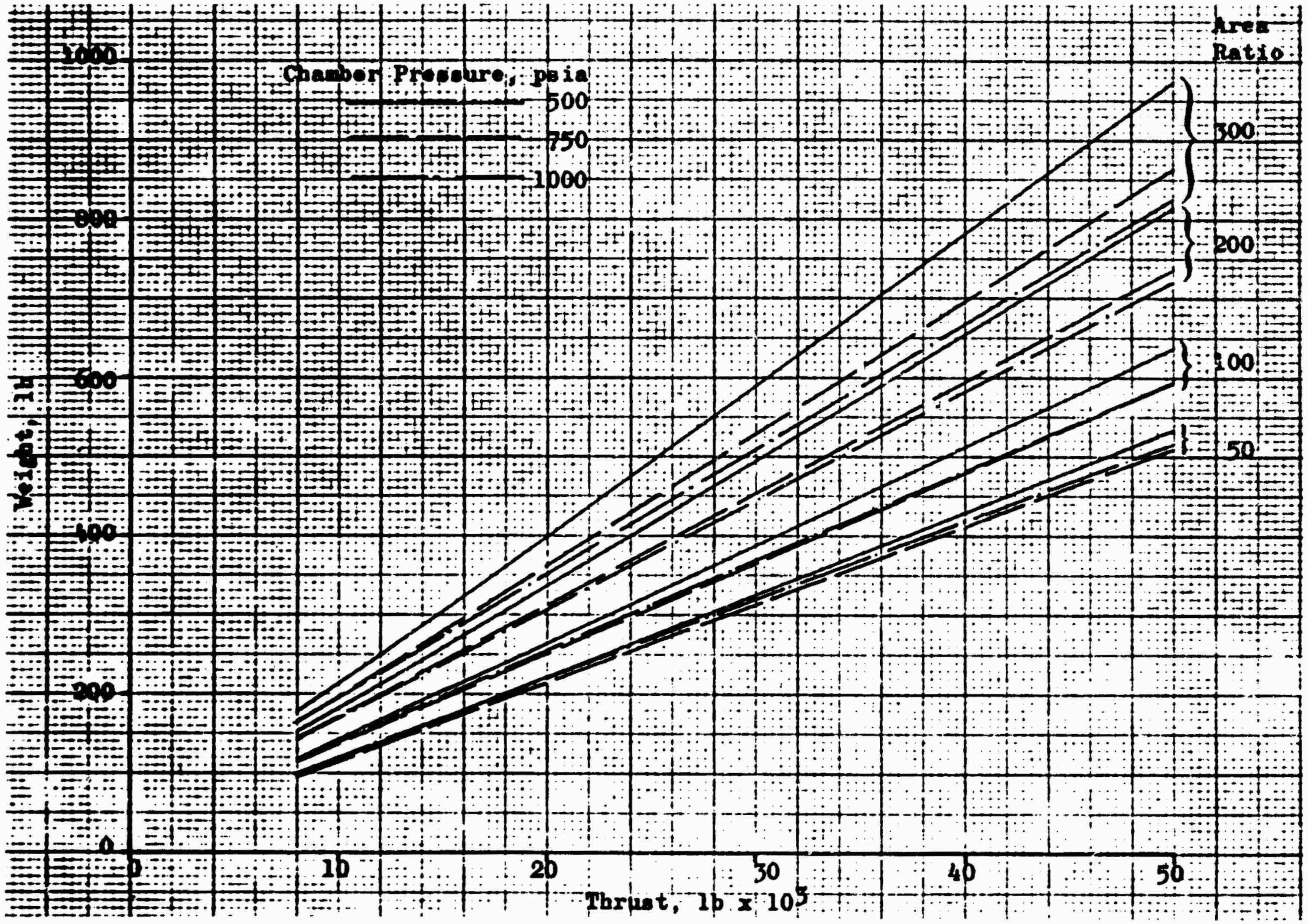


Figure 180. Aerospike Engine System Dry Weight for Expander Topping Cycle, Engine Mixture Ratio = 5.5

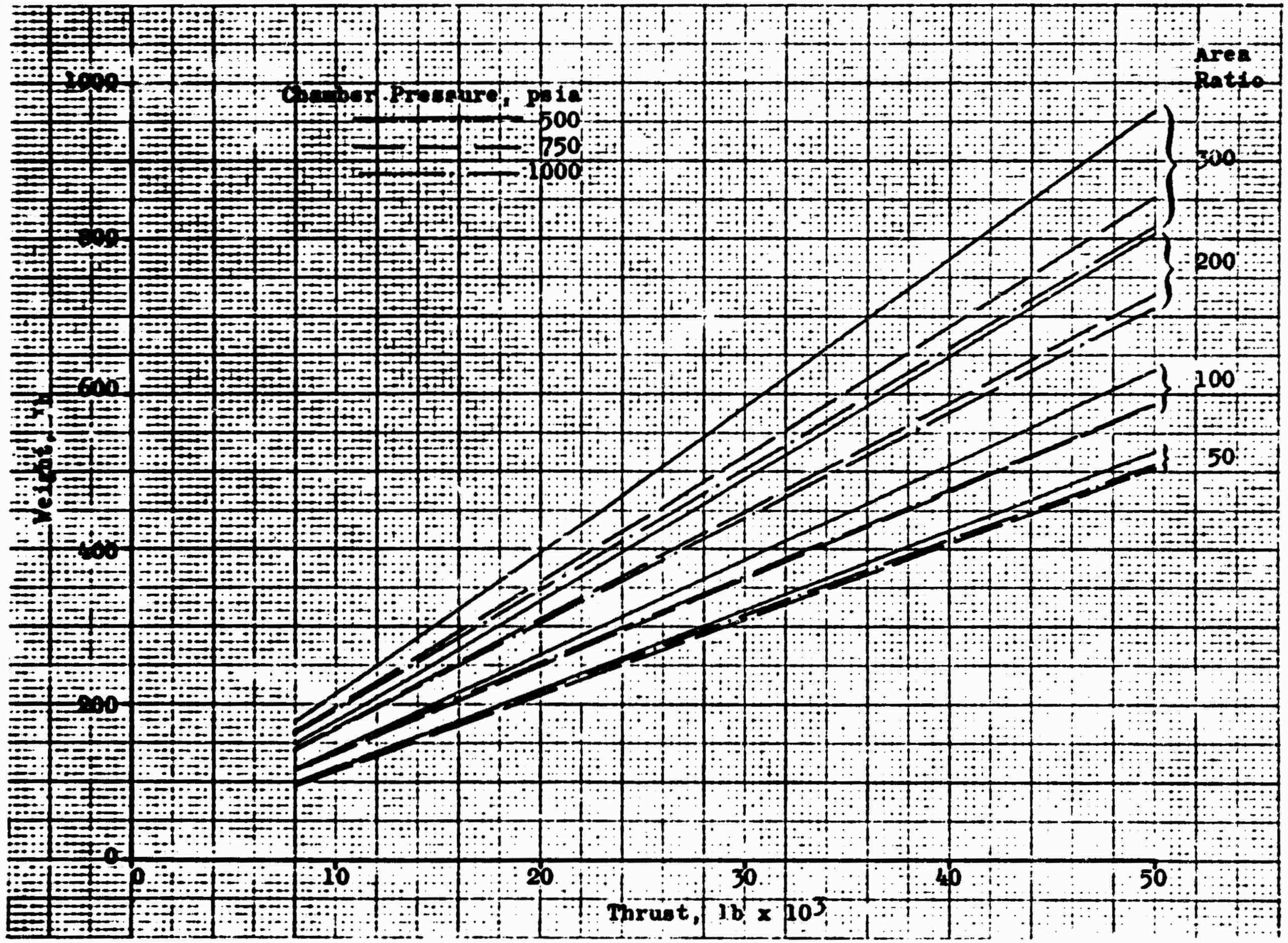


Figure 181. Aerospike Engine System Dry Weight for Expander Topping Cycle, Engine Mixture Ratio = 6.0

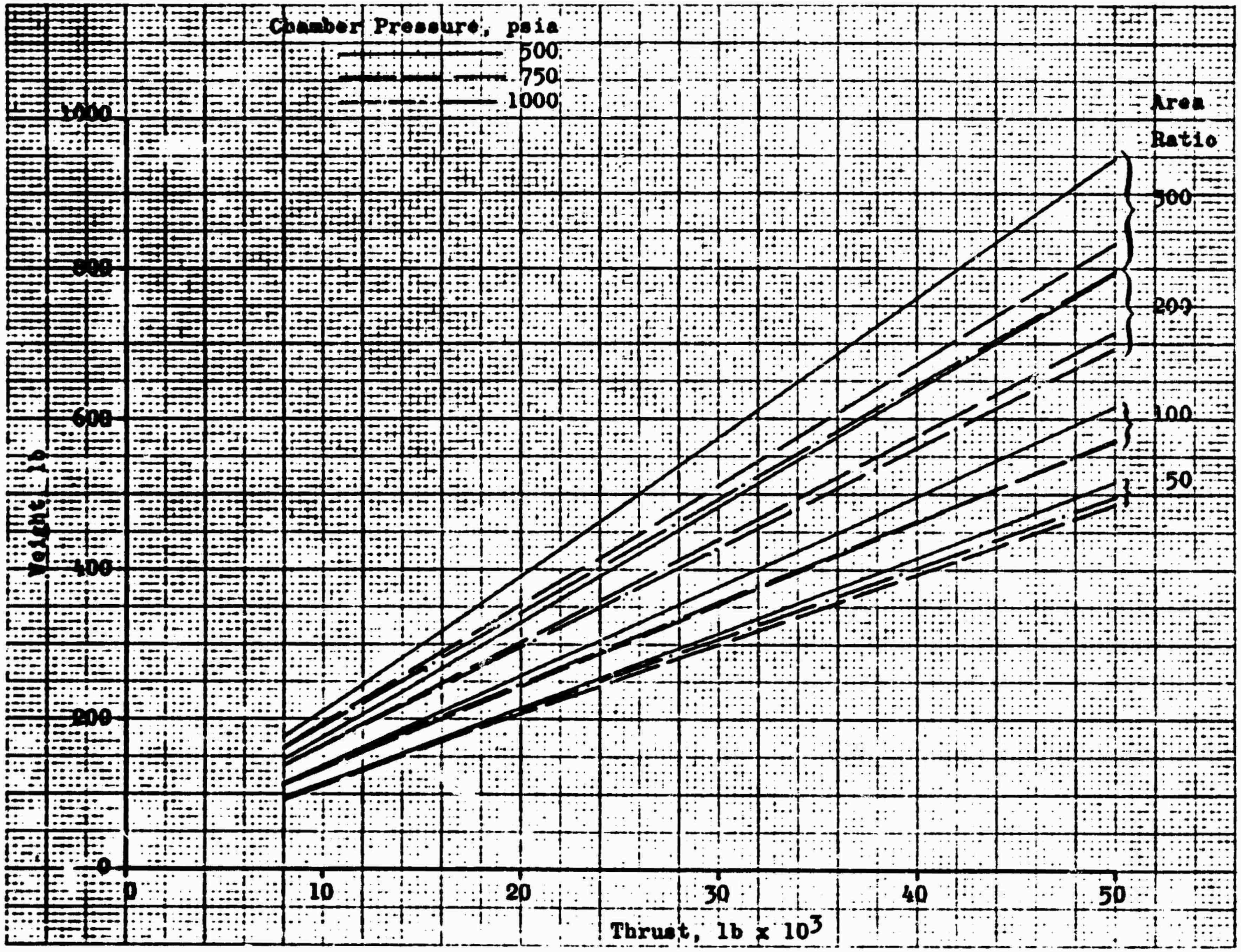


Figure 182. Aerospike Engine System Dry Weight for Expander Topping Cycle, Engine Mixture Ratio = 7.0

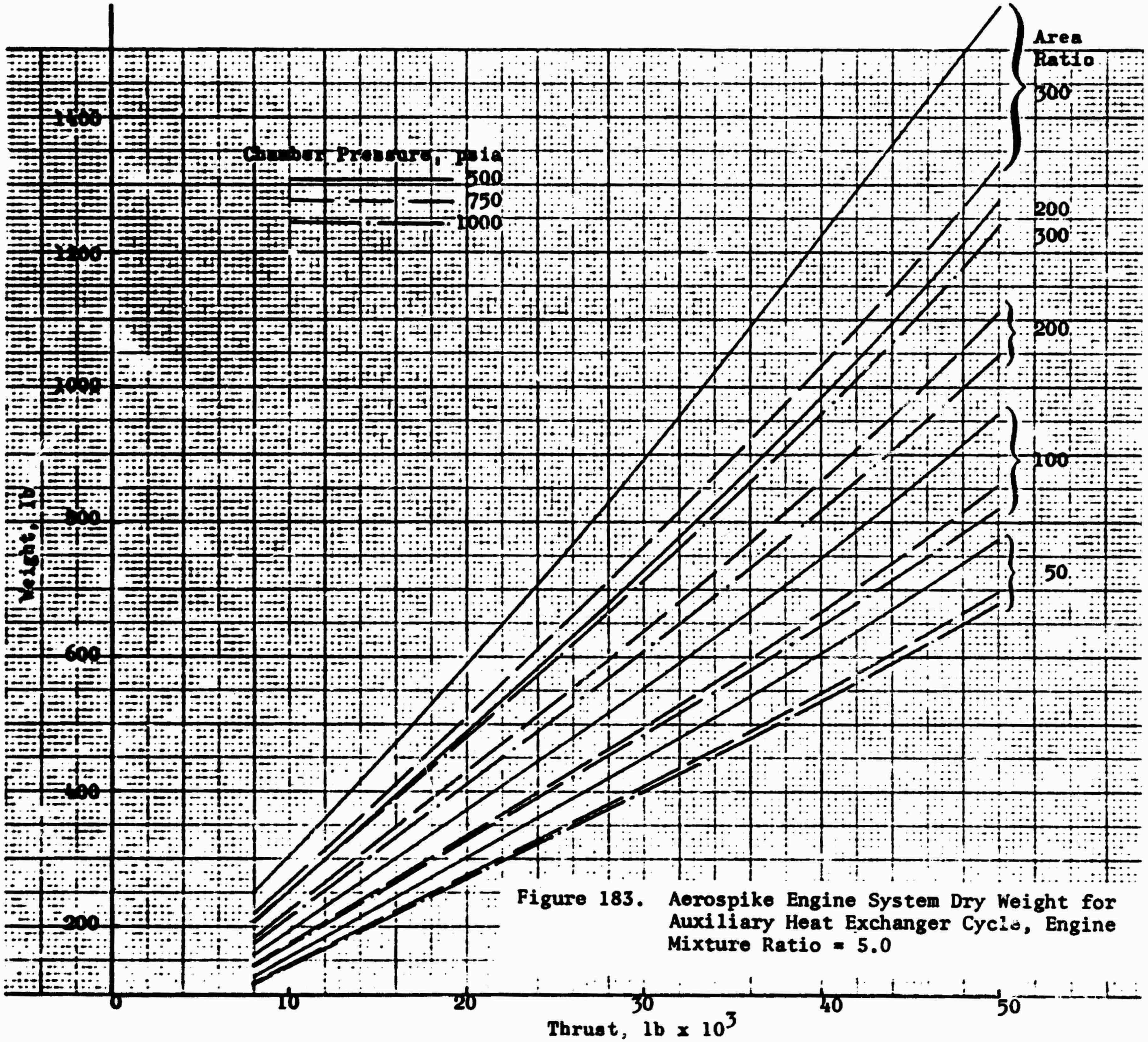


Figure 183. Aerospike Engine System Dry Weight for Auxiliary Heat Exchanger Cycle, Engine Mixture Ratio = 5.0

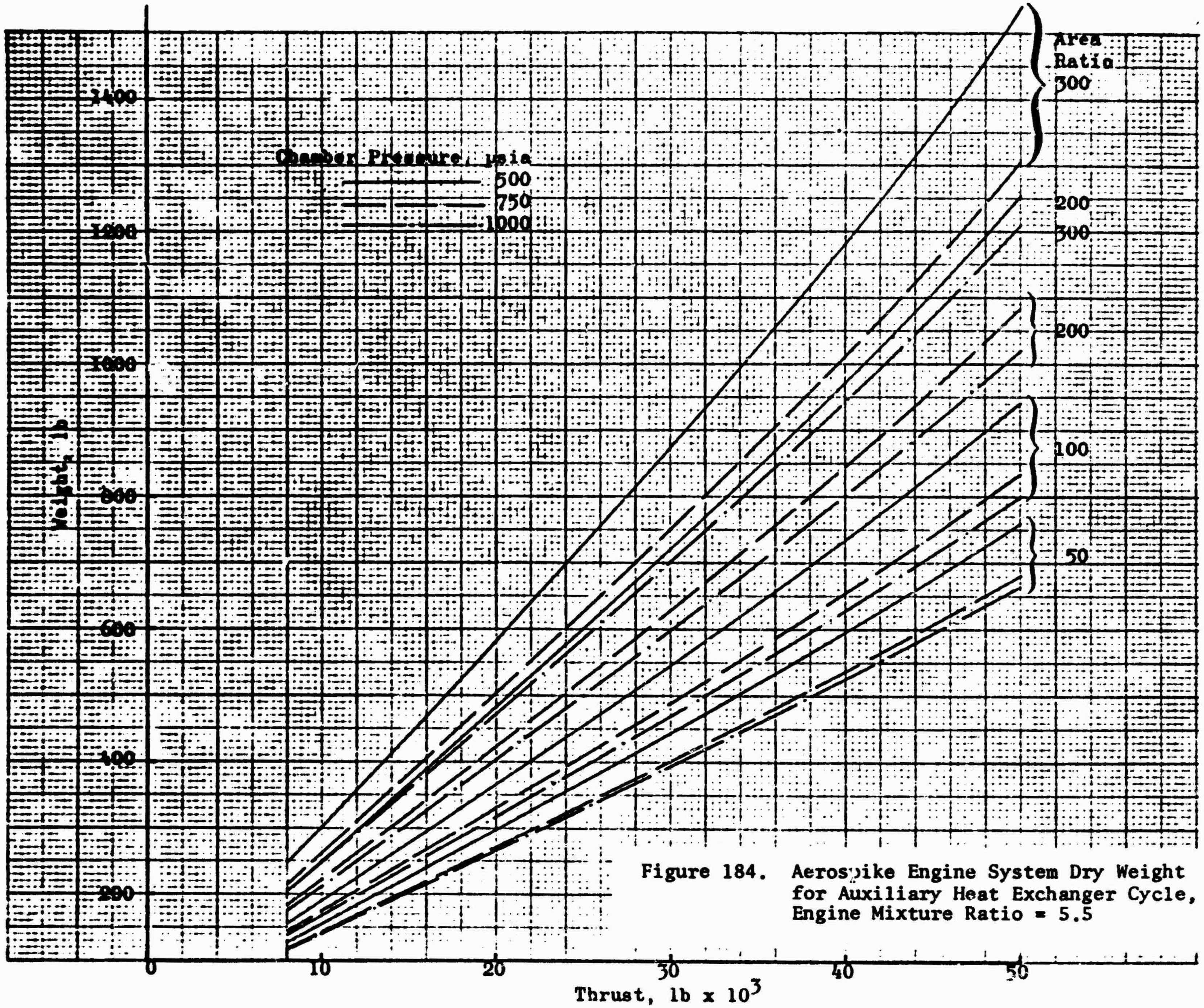


Figure 184. Aerospike Engine System Dry Weight for Auxiliary Heat Exchanger Cycle, Engine Mixture Ratio = 5.5

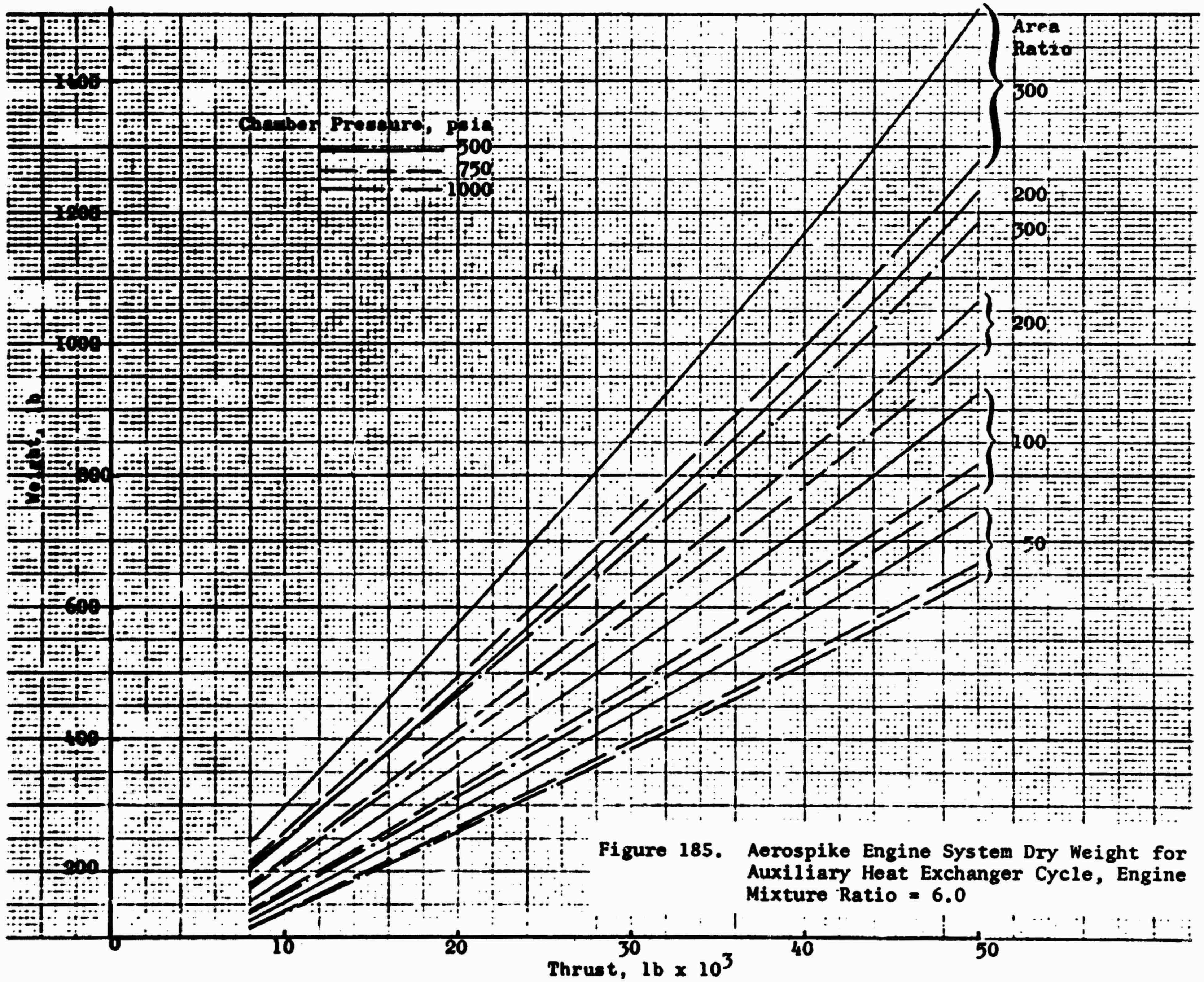


Figure 185. Aerospike Engine System Dry Weight for Auxiliary Heat Exchanger Cycle, Engine Mixture Ratio = 6.0

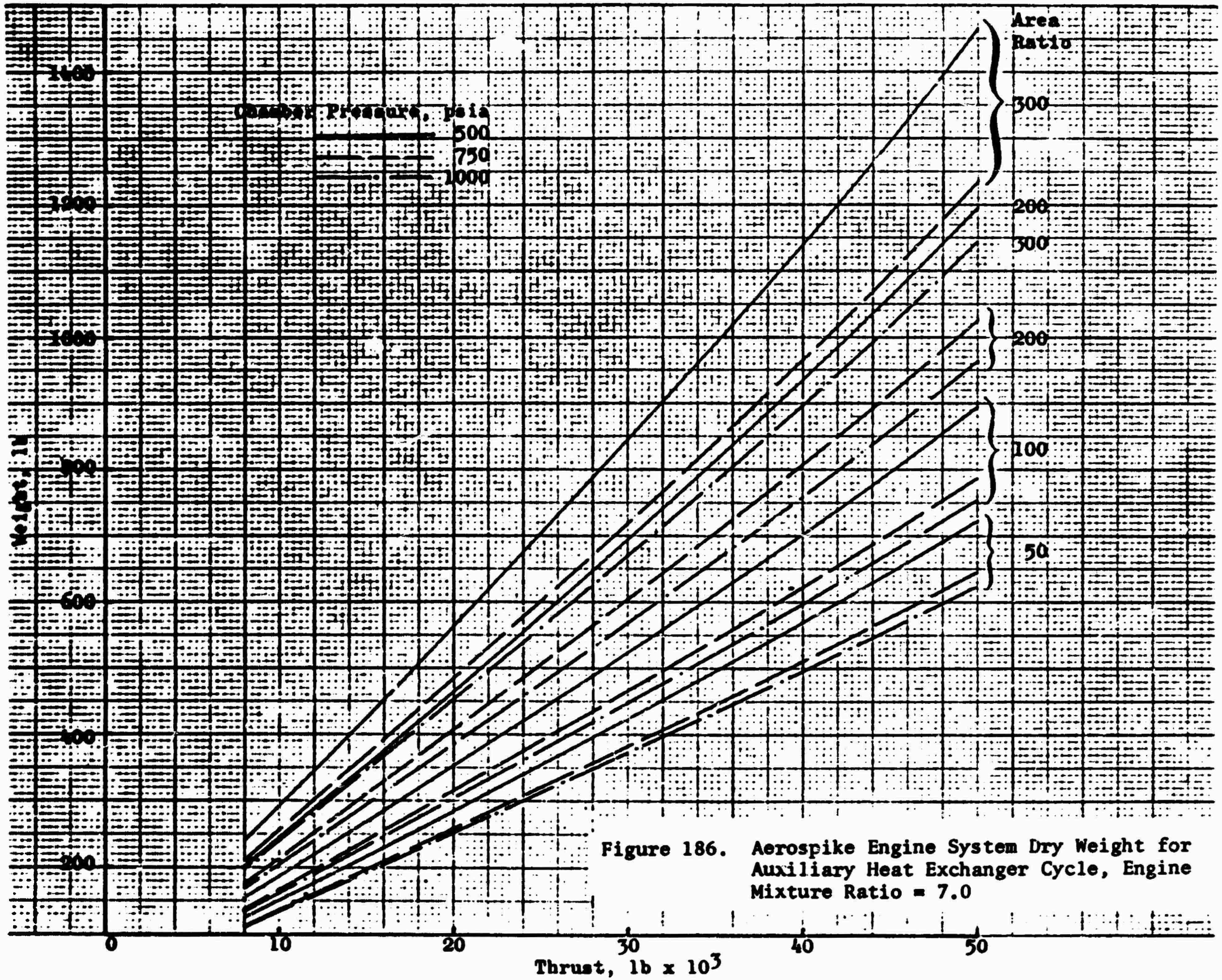


Figure 186. Aerospike Engine System Dry Weight for Auxiliary Heat Exchanger Cycle, Engine Mixture Ratio = 7.0

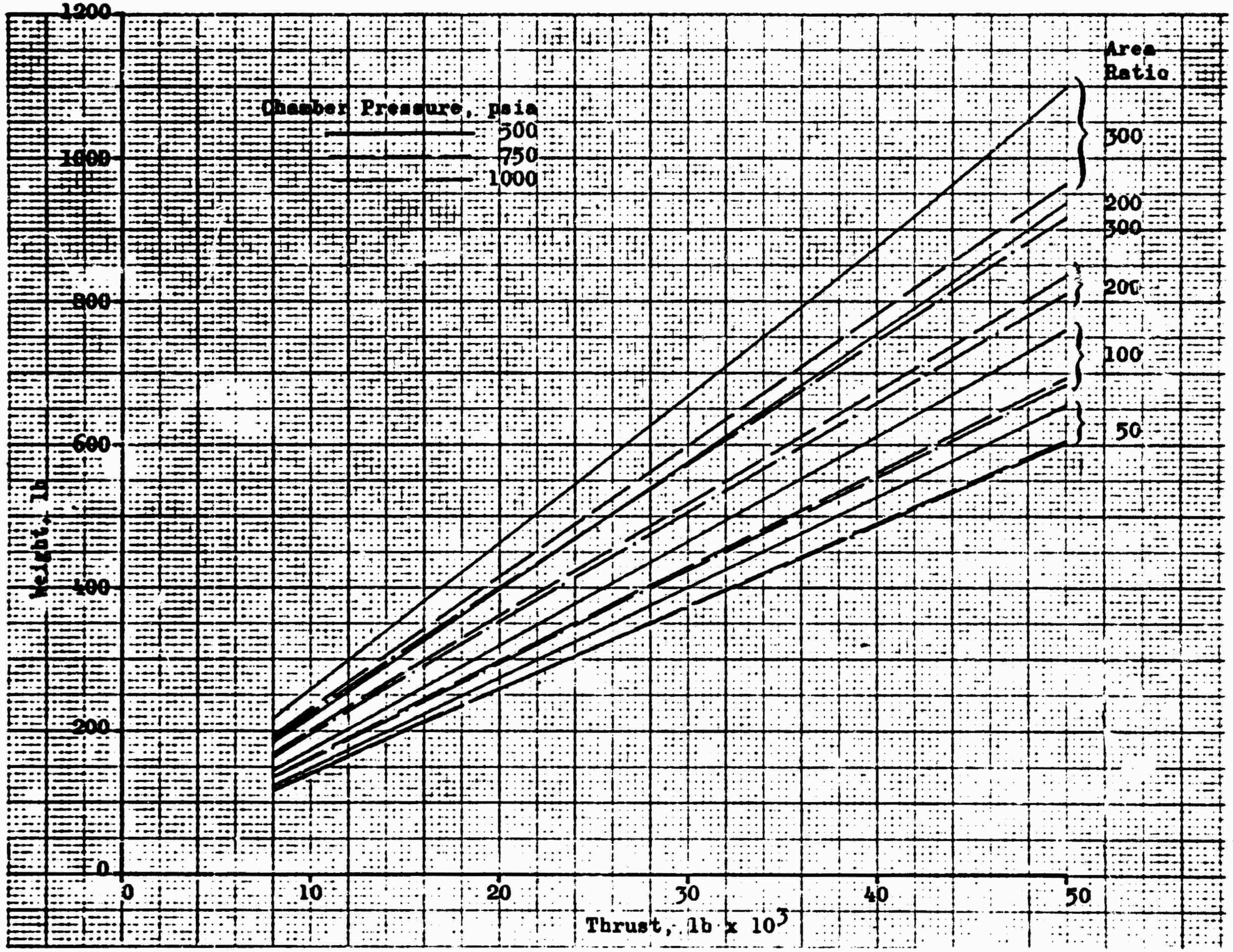


Figure 187. Aerospike Engine System Dry Weight for Gas Generator Cycle, Engine Mixture Ratio = 5.0

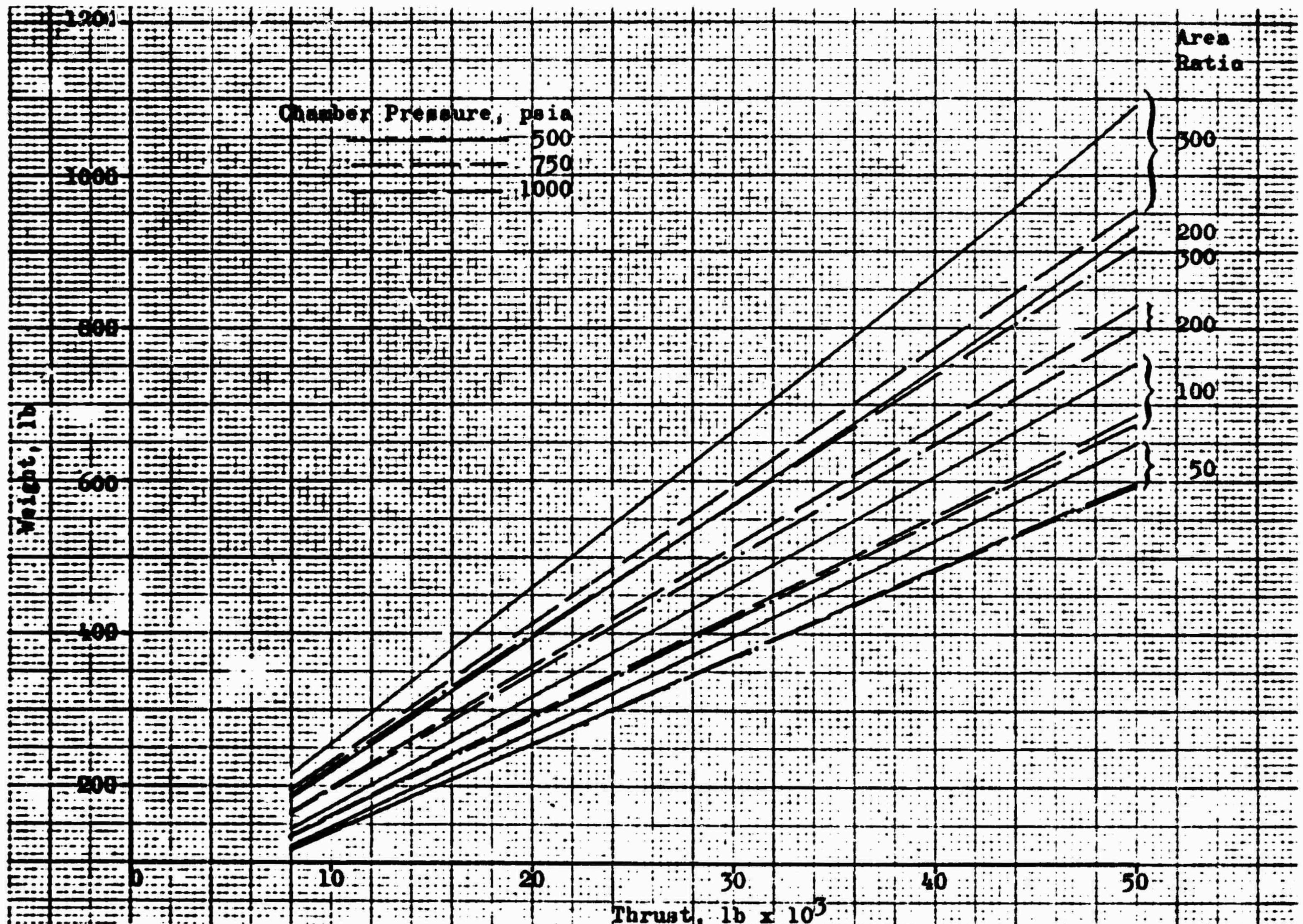


Figure 188. Aerospike Engine System Dry Weight for Gas Generator Cycle, Engine Mixture Ratio = 5.5

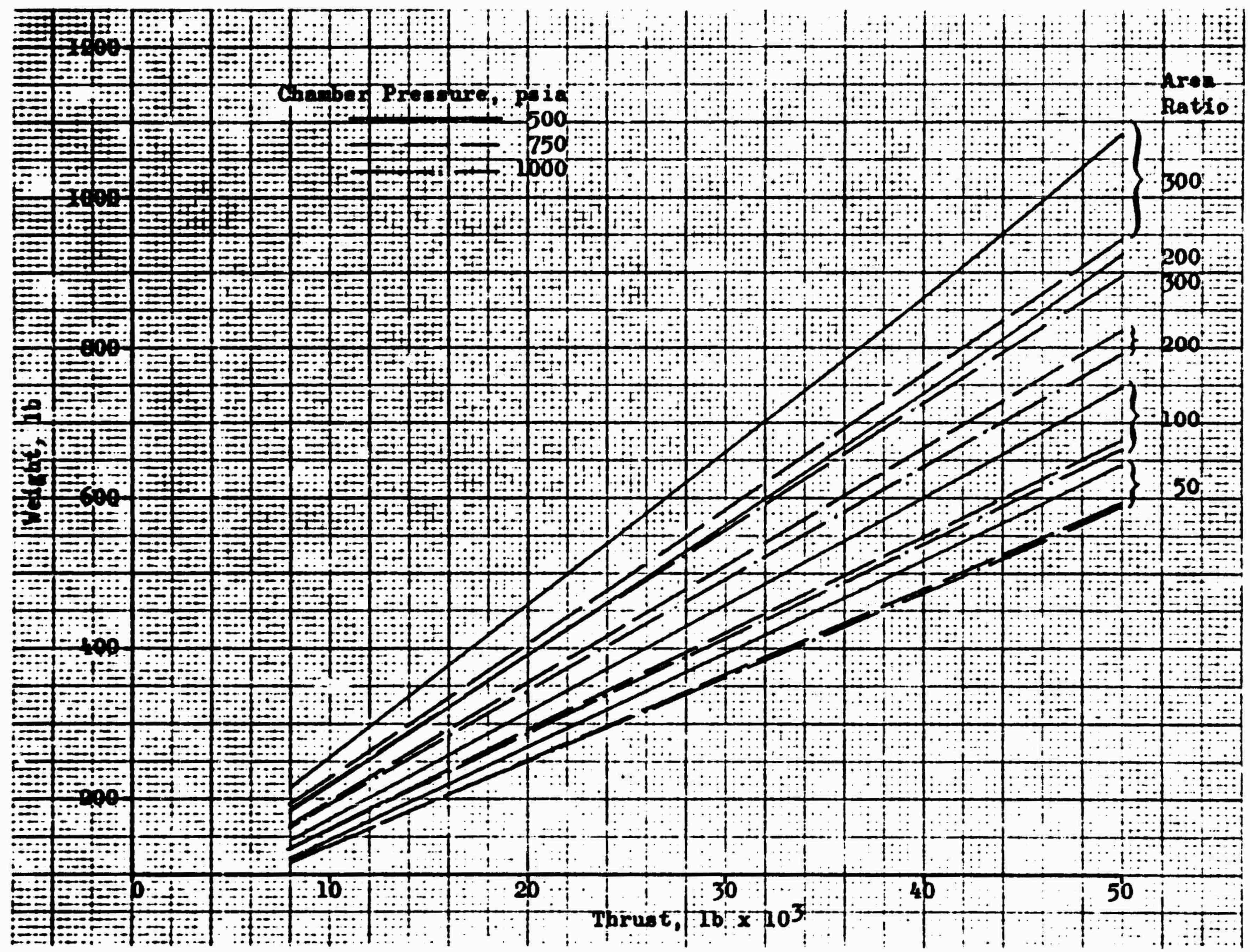


Figure 189. Aerospike Engine System Dry Weight for Gas Generator Cycle, Engine Mixture Ratio = 6.0

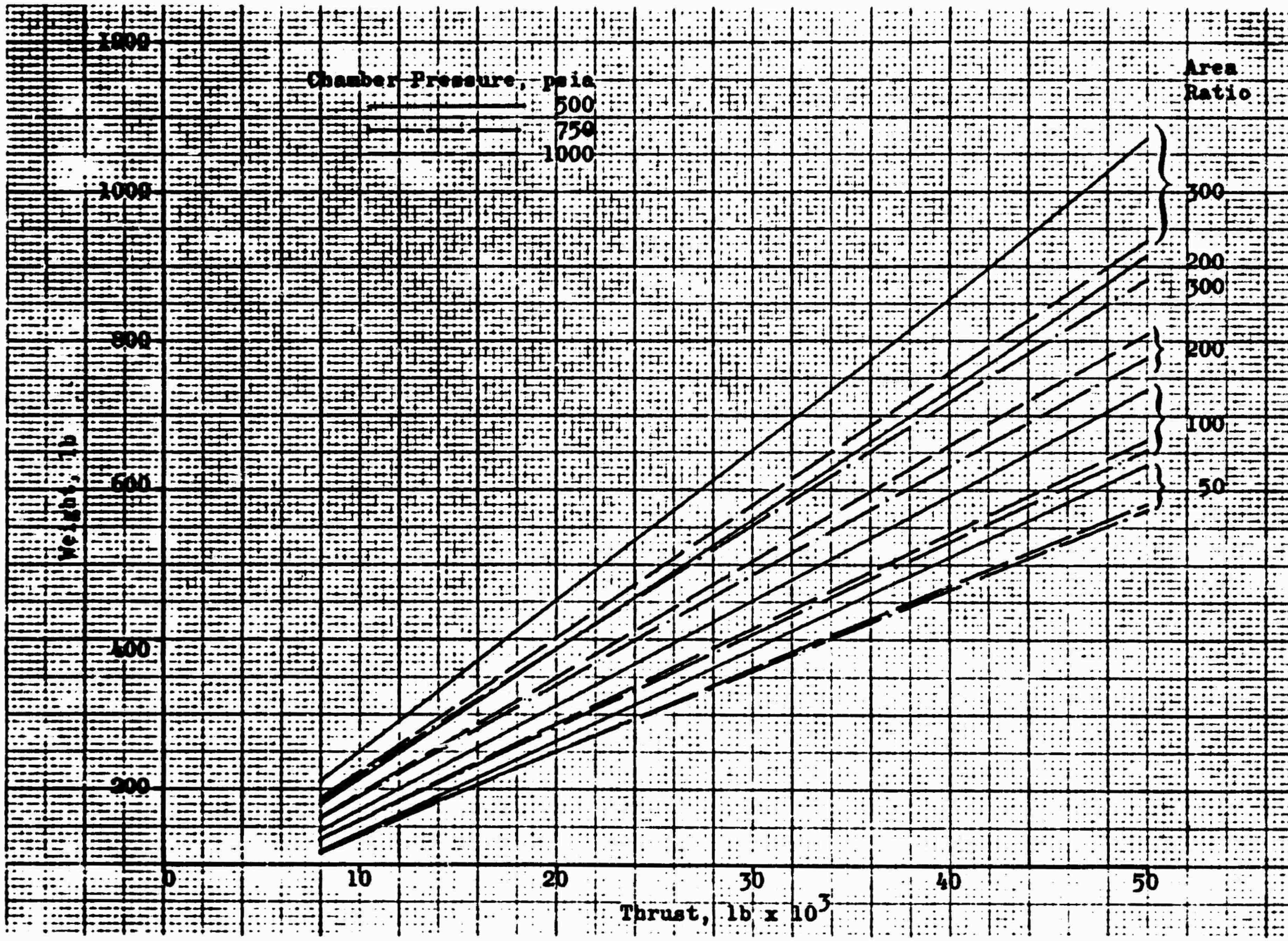


Figure 190. Aerospike Engine System Dry Weight for Gas Generator Cycle,
Engine Mixture Ratio = 7.0

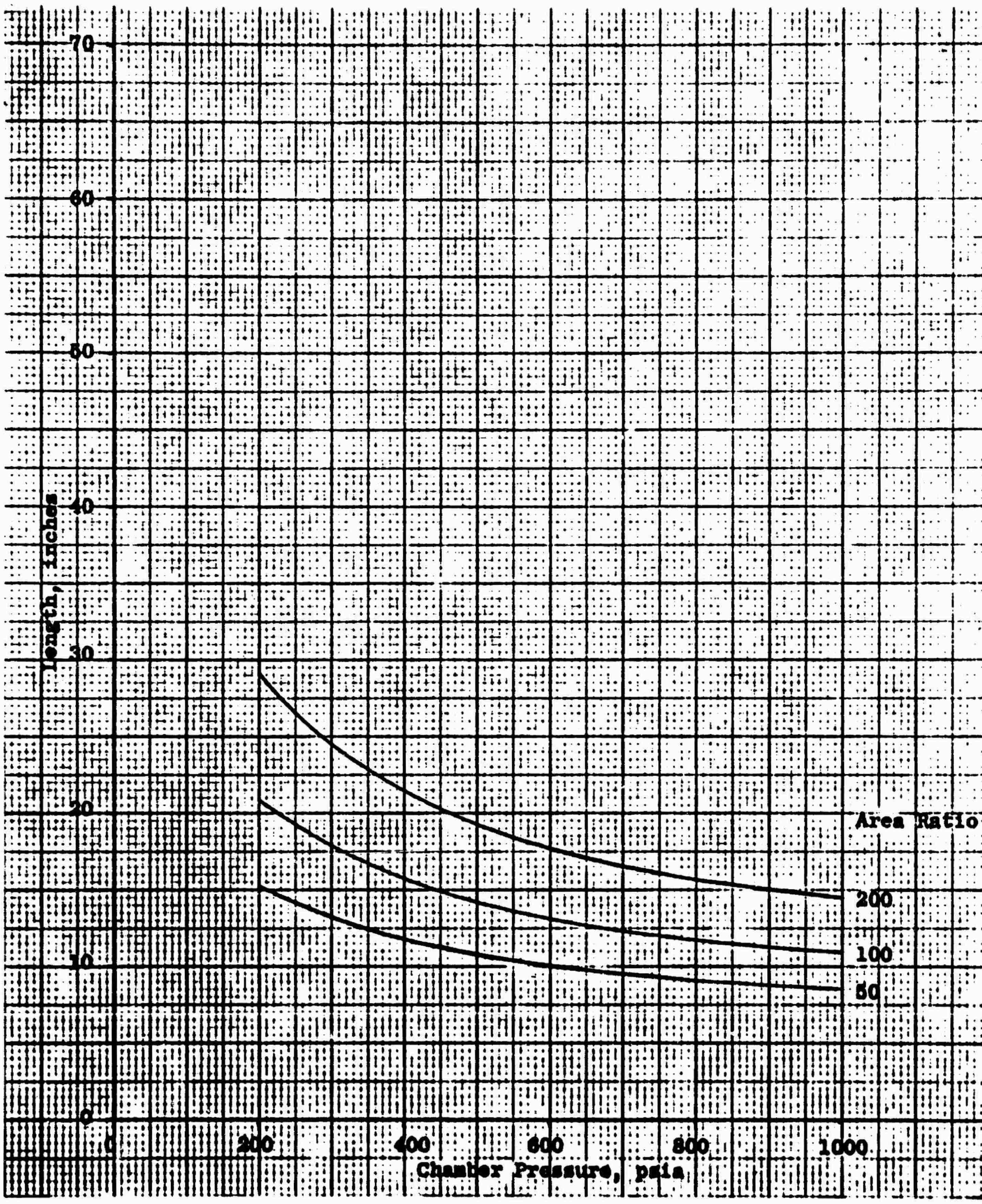


Figure 191. Aerospike Engine Length for 8000-Pound Thrust

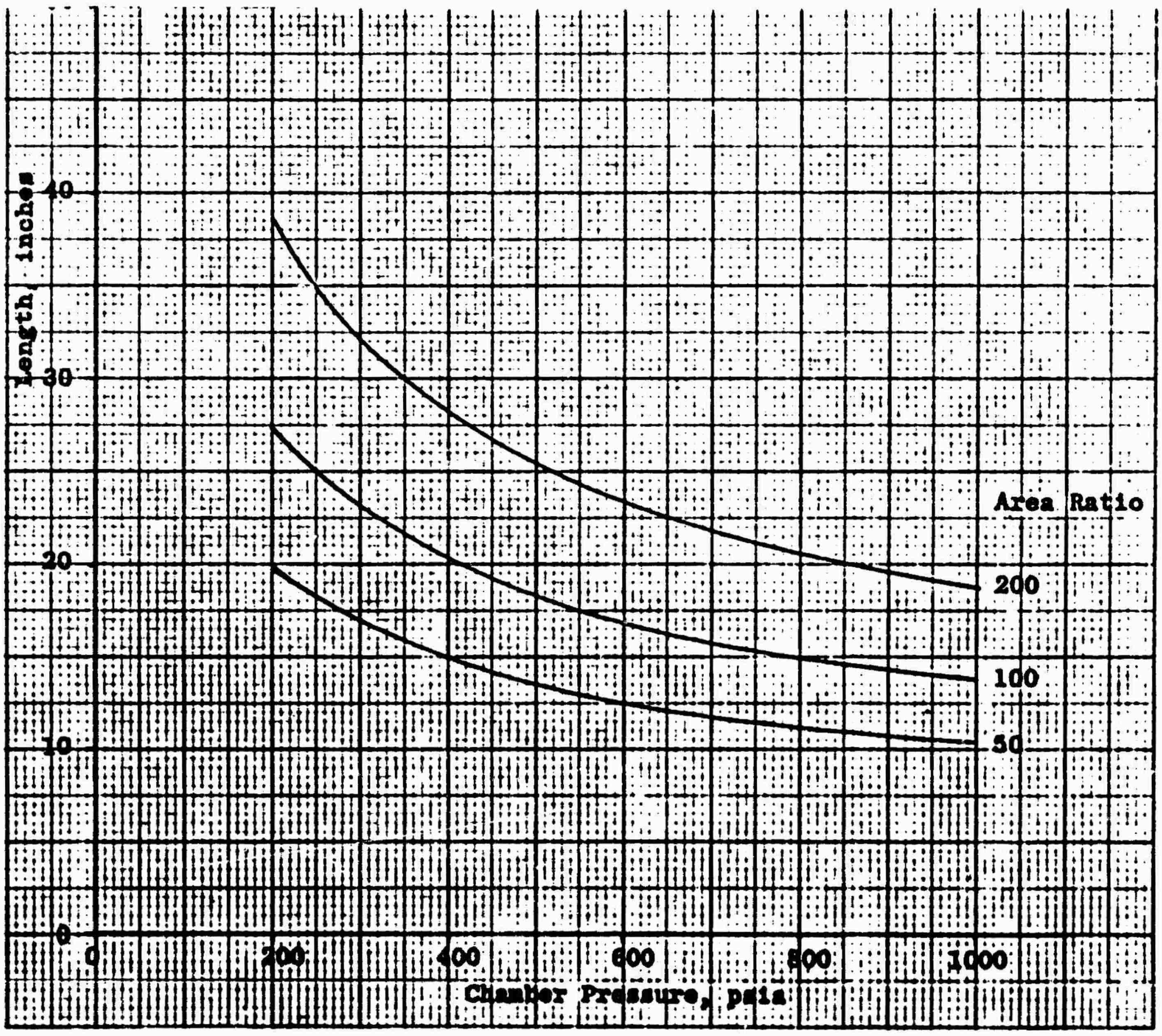


Figure 192. Aerospike Engine Length for 15,000-Pound Thrust

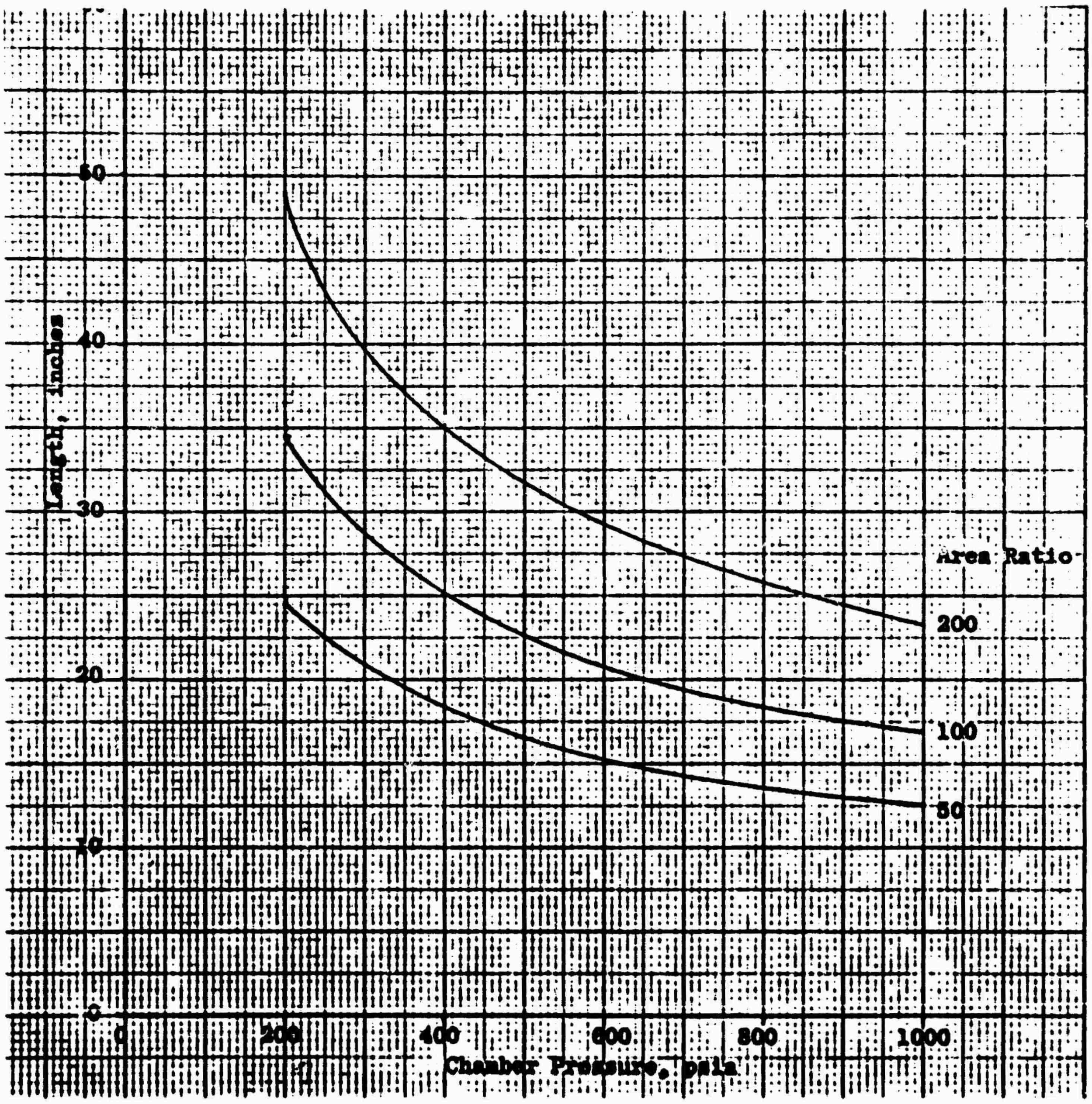


Figure 193. Aerospike Engine Length for 25,000-Pound Thrust

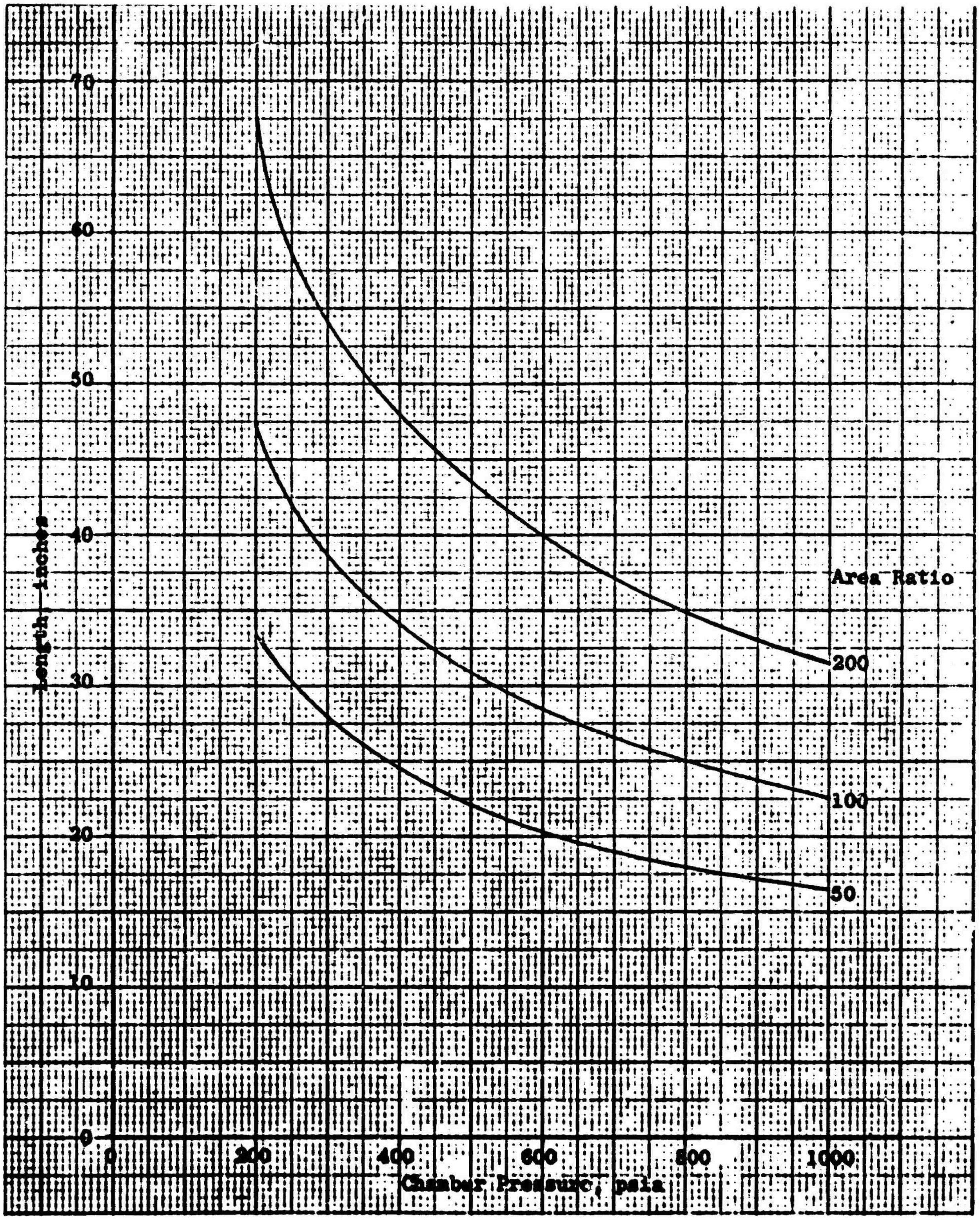


Figure 194. Aerospike Engine Length for 50,000-Pound Thrust

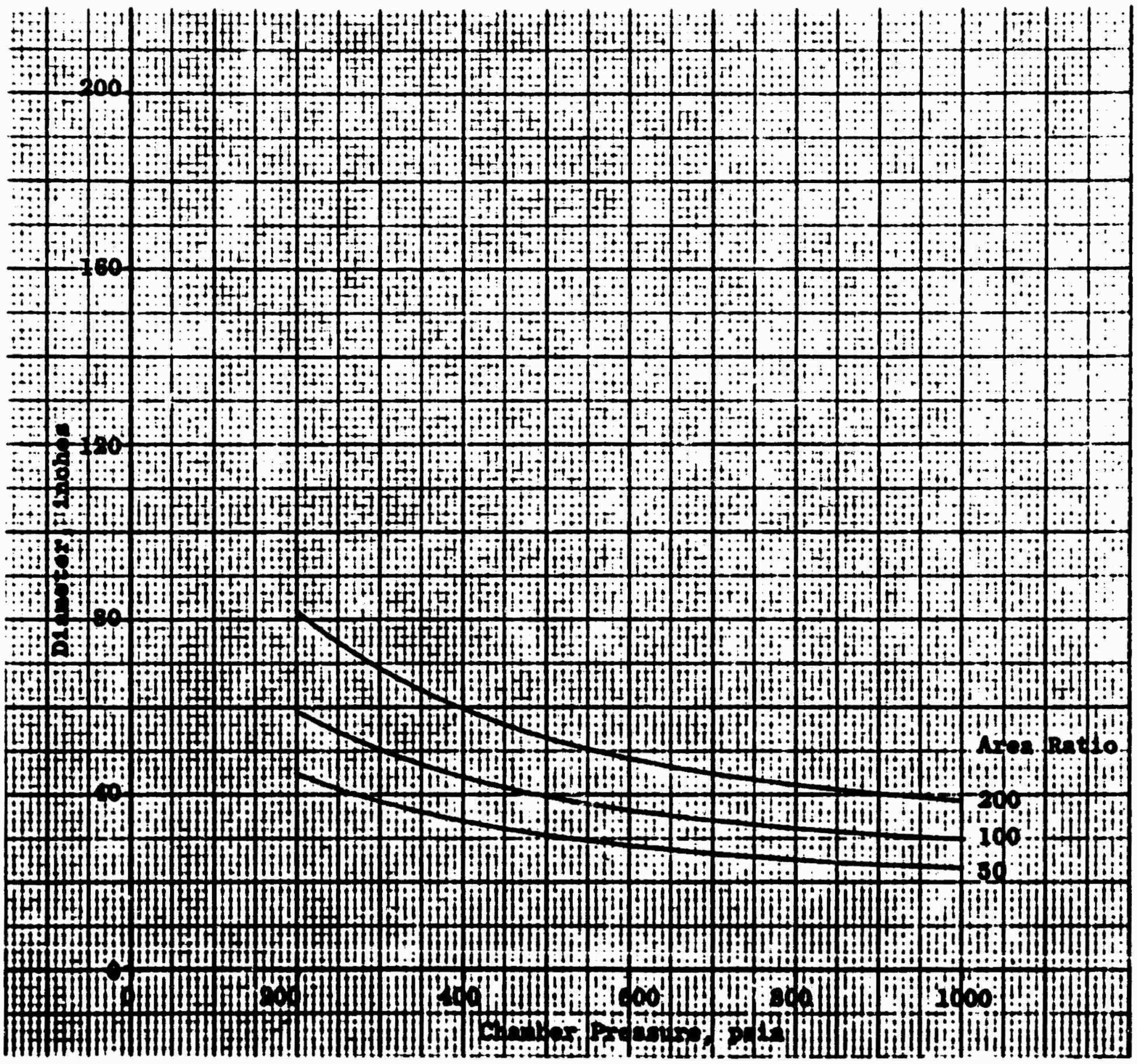


Figure 195. Aerospike Engine Diameter for 8000-Pound Thrust

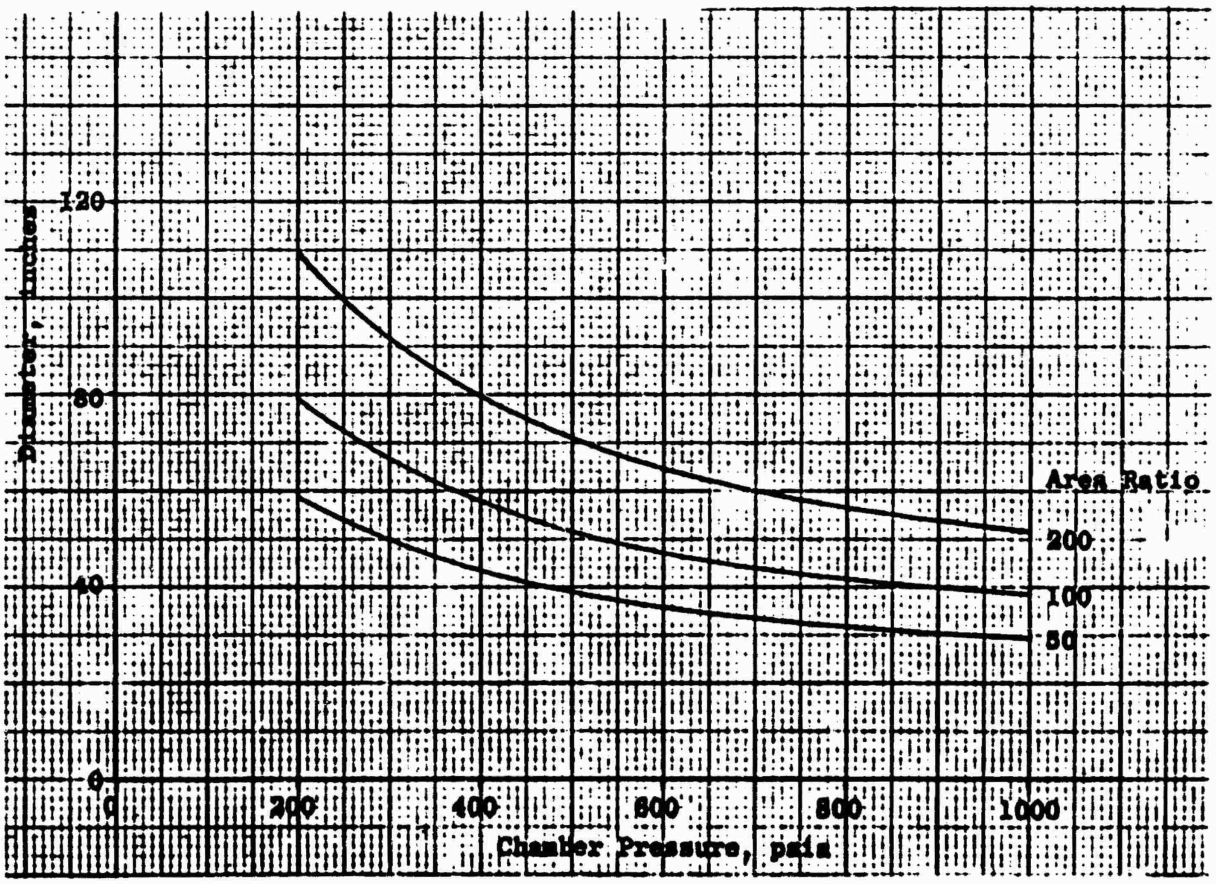


Figure 196. Aerospike Engine Diameter for 15,000-Pound Thrust

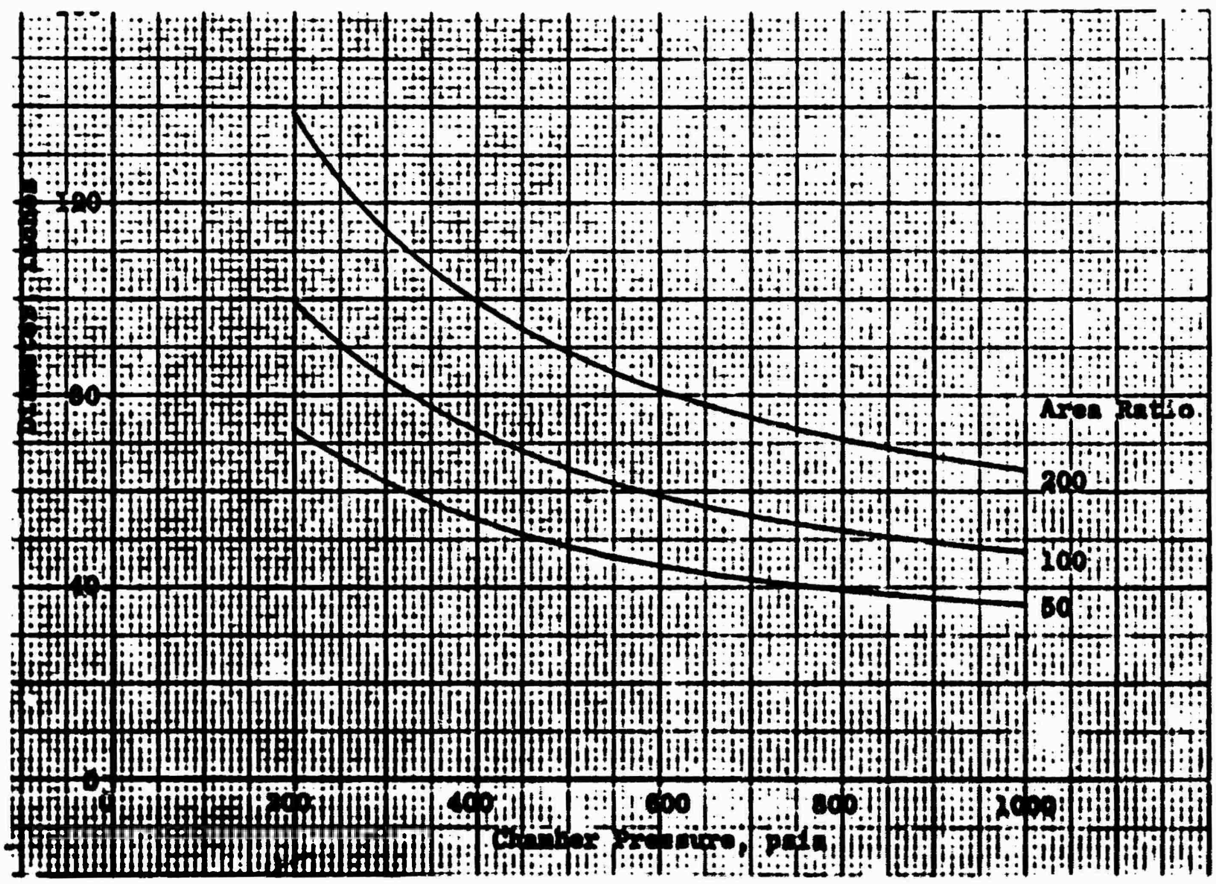


Figure 197. Aerospike Engine Diameter for 25,000-Pound Thrust

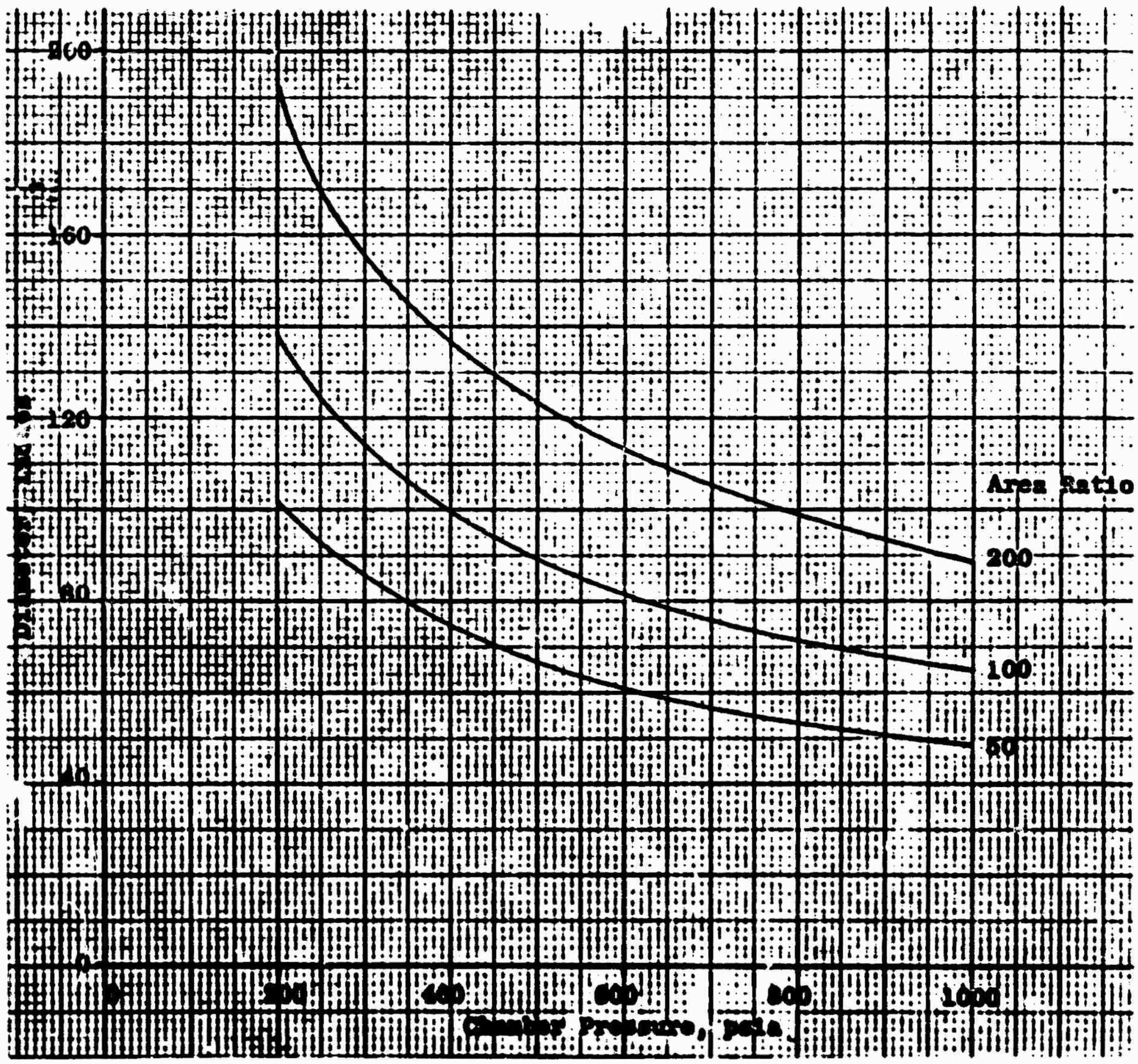


Figure 198. Aerospike Engine Diameter for 50,000-Pound Thrust

ENGINE ELECTRICAL AND PNEUMATIC REQUIREMENTS

This section describes those components requiring electrical power. Electrical power and energy storage requirements are specified. Engine system pneumatic fluid requirements also are specified.

ENGINE CONFIGURATION OPTIMIZATION

The parametric engine performance and weight information generated was used as a basis for a tradeoff study to establish the optimum engine design point for the maximum thrust levels of 8,000, 15,000, 25,000, and 50,000 pounds. Both single-panel and double-panel aerospike cooling circuits were considered.

For a fixed engine mixture ratic, the payload capability of an engine in a given mission depends on its delivered specific impulse and weight. Because specific impulse can quite often be purchased at the expense of weight and because the exchange factors on specific impulse and weight are seldom the same (e.g., in a high-energy mission such as the low earth orbit to synchronous orbit mission, specific impulse in highly favored in relation to weight), the optimum engine configuration is not necessarily the lowest weight and/or the highest specific impulse configuration.

To facilitate determination of the optimum configuration, the parametric engine weight and performance data presented in the aerospike parametric information report may be replotted as specific impulse versus engine dry weight for a given thrust level, mixture ratio, and cycle. If cooling limits and (for closed cycles) power limits are superimposed on this plot, a graph showing the feasible area of operation for the cycle at the given thrust level and engine mixture ratio results. This graph can be used to determine the optimum engine configuration for any mission by plotting straight lines having as their slope the ratio of the mission engine weight exchange factor to the mission specific impulse exchange factor, i.e.,

$$\left\{\frac{\Delta PL}{\Delta We}\right\} \Big/ \left\{\frac{\Delta PL}{\Delta I_s}\right\} = \left\{\frac{\Delta I_s}{\Delta We}\right\}$$

Each line represents constant payload for the given mission. Payload is increased by moving upwards and to the left in a direction perpendicular to the straight lines. The point of tangency between the constant payload line and the uppermost and left-most point in the feasible region of operation represents the optimum engine configuration.

Figure 159 presents the single- and double-panel aerospike regions of operation at 8000 pounds thrust and a mixture ratio of 5.5:1 for both expander cycles and gas generator cycles. The single-panel expander is limited to low area ratios, due to the high heat flux and low hydrogen flowrate occurring at 8000 pounds thrust, and to low chamber pressures ($\sim$650 psia) due to the high coolant jacket pressure drops required to maintain a reasonable wall temperature. That is, the power limit occurs at a relatively low chamber pressure. The figure also indicates that the gas generator cycle significantly underperforms the expander cycle. This performance difference is characteristic of the two cycles and becomes more marked as thrust increases; thus, only expander cycles are shown at thrust level of 25,000 and 50,000 pounds. The double-panel aerospike has a very limited region of operation at very low thrusts such as 8000 pounds due to the cooling circuit, which involves a double pass of the hydrogen through the throat region which requires extremely high pressure drops to maintain resonable wall temperatures at low thrusts when near the cooling limit. Therefore, the full potential of the double-panel concept cannot be realized until higher thrust levels are reached (by 15,000 pounds thrust, the pressure drops are down and the double-panel engine has a very reasonable region of operation)

A feasible double-panel design and the optimum single-panel configuration are shown by circles in Fig. 199.

The feasible regions of operation for both the single- and the double-panel expander cycle aerospikes at 15,000 pounds thrust and an engine mixture ratio of 5.5:1 are shown in Fig. 200. For comparison, a double-panel gas generator cycle is shown in Fig. 201. The two figures show that, for the double-panel aerospike (as for the single panel), the gas generator cycle significantly

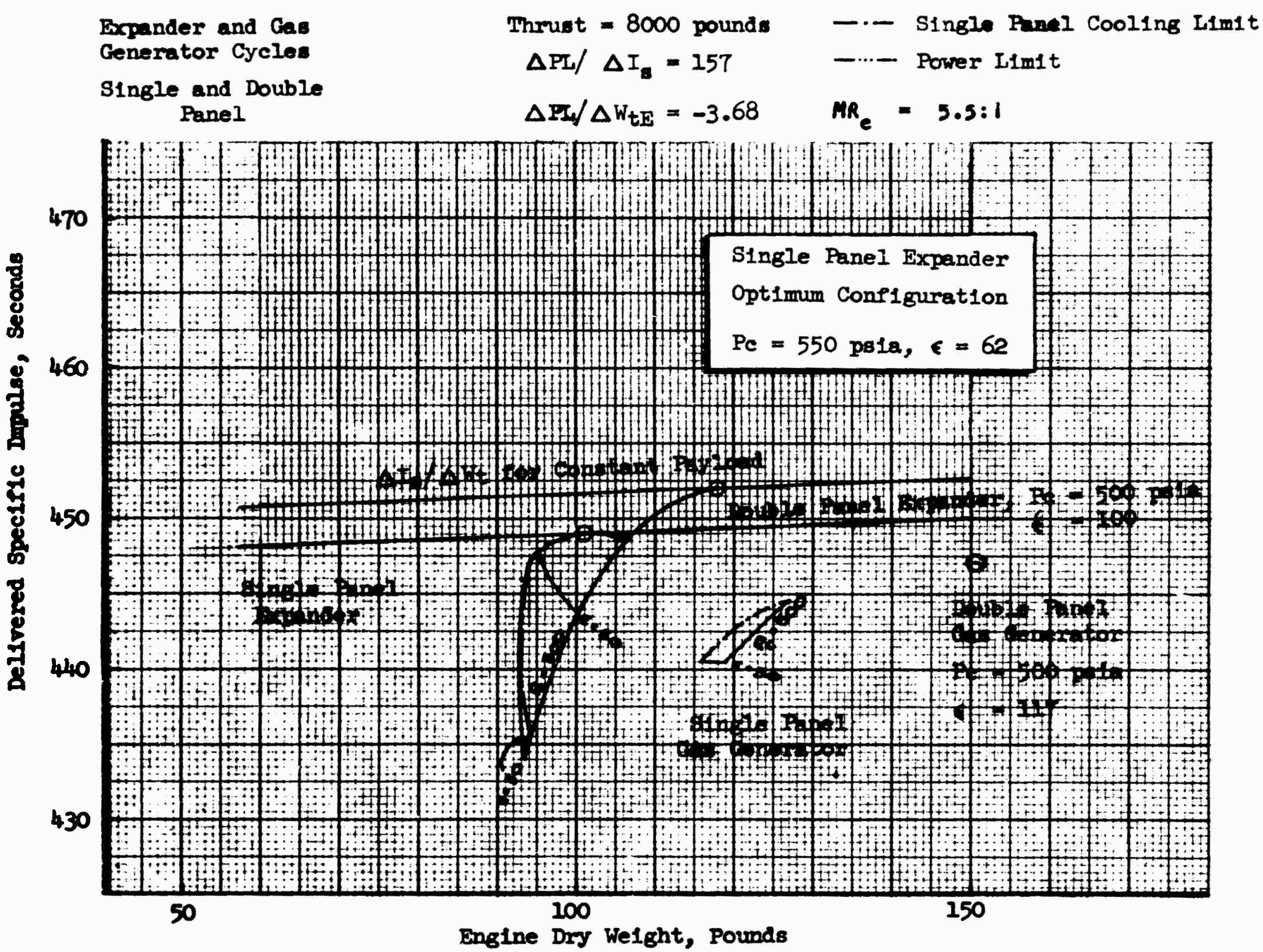


Figure 199. Aerospike Engine Optimization

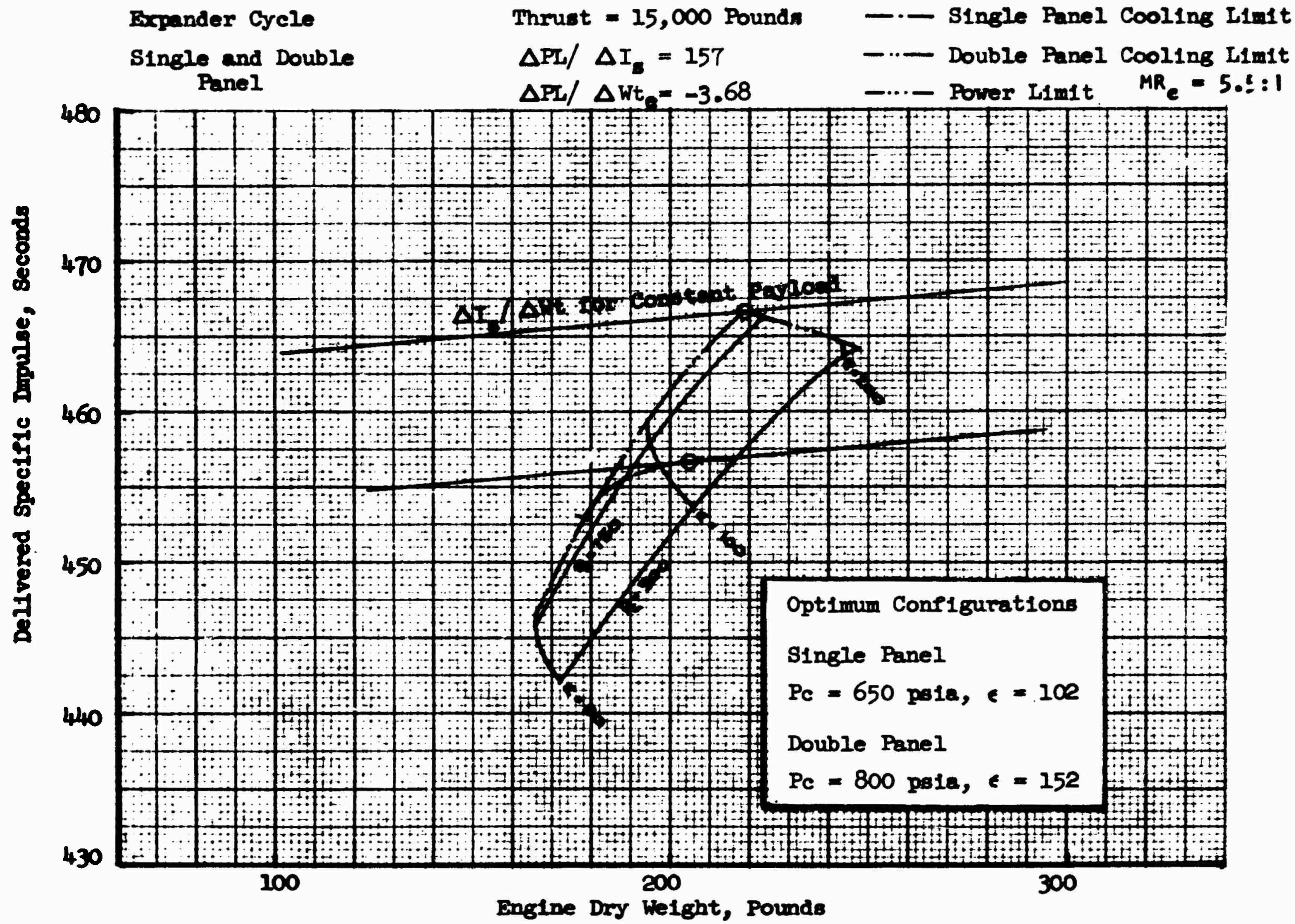


Figure 200. Aerospike Engine Optimization

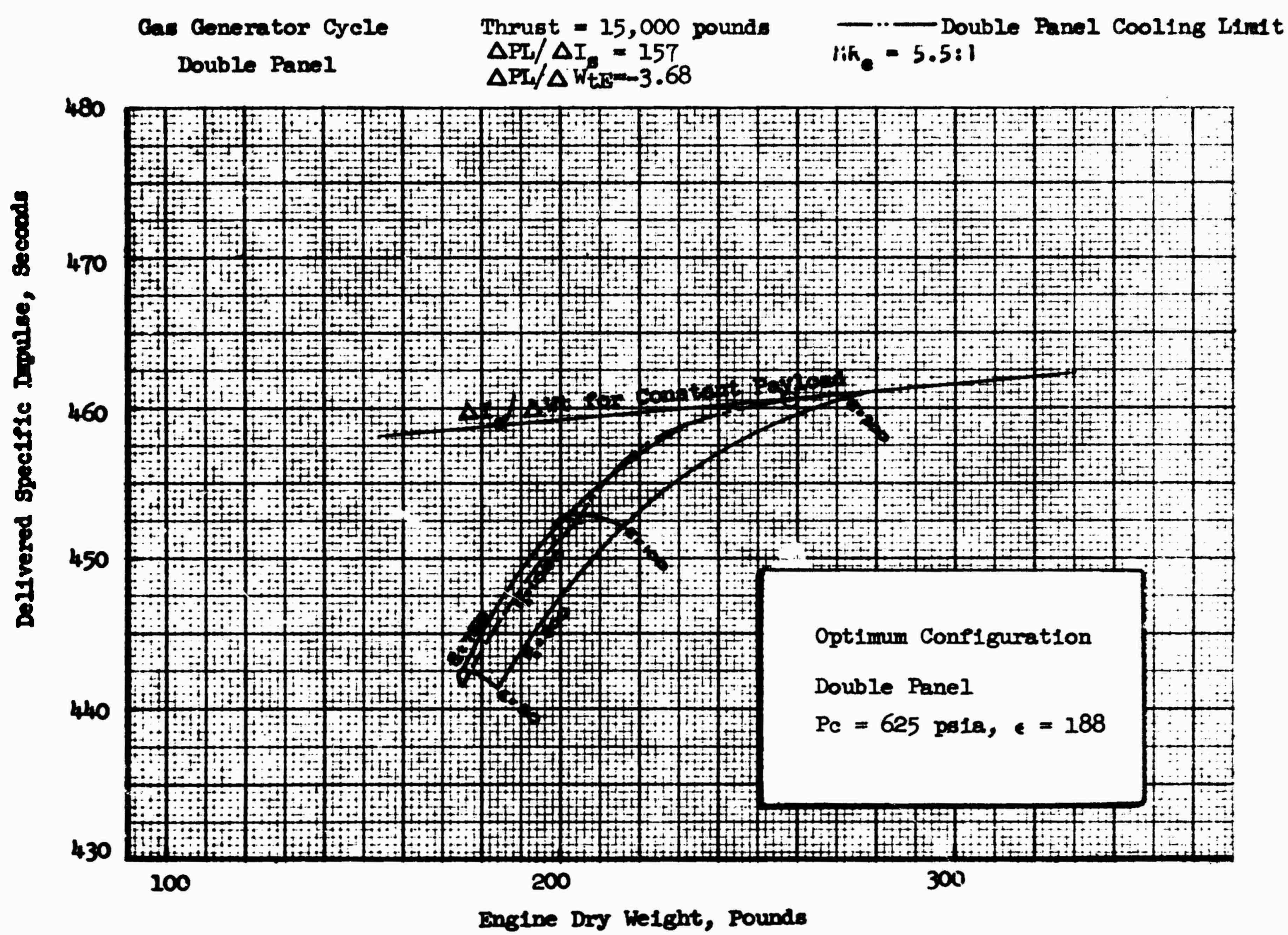


Figure 201. Aerospike Engine Optimization

underperforms the expander cycle and, thus, for the higher thrust, the double-panel gas generator will not be shown. Optimum configurations are shown by a circle in Fig. 200 and 201, and chamber pressures and area ratios for the optimum design points also are indicated.

Figures 202 and 203 present the single- and double-panel expanders at thrusts of 25,000 and 50,000 pounds, respectively, and both at 5.5:1 mixture ratio. Optimums are indicated as previously.

Table 80 summarizes the single- and double-panel aerospike optimum configurations at 8,000 pounds, 15,000 pounds, 25,000 pounds, and 50,000 pounds thrust. Selected parameters are shown as a function of thrust in Fig. 204 and 205 .

The optimum single-panel, 25,000-pound-thrust engine system design point does not correspond to the baseline single-panel engine system specified by the contract work statement and described in this report because of a relaxation of the thrust chamber cooling limit which was made possible through refinements in the regenerative coolant system design parameters and circuitry.

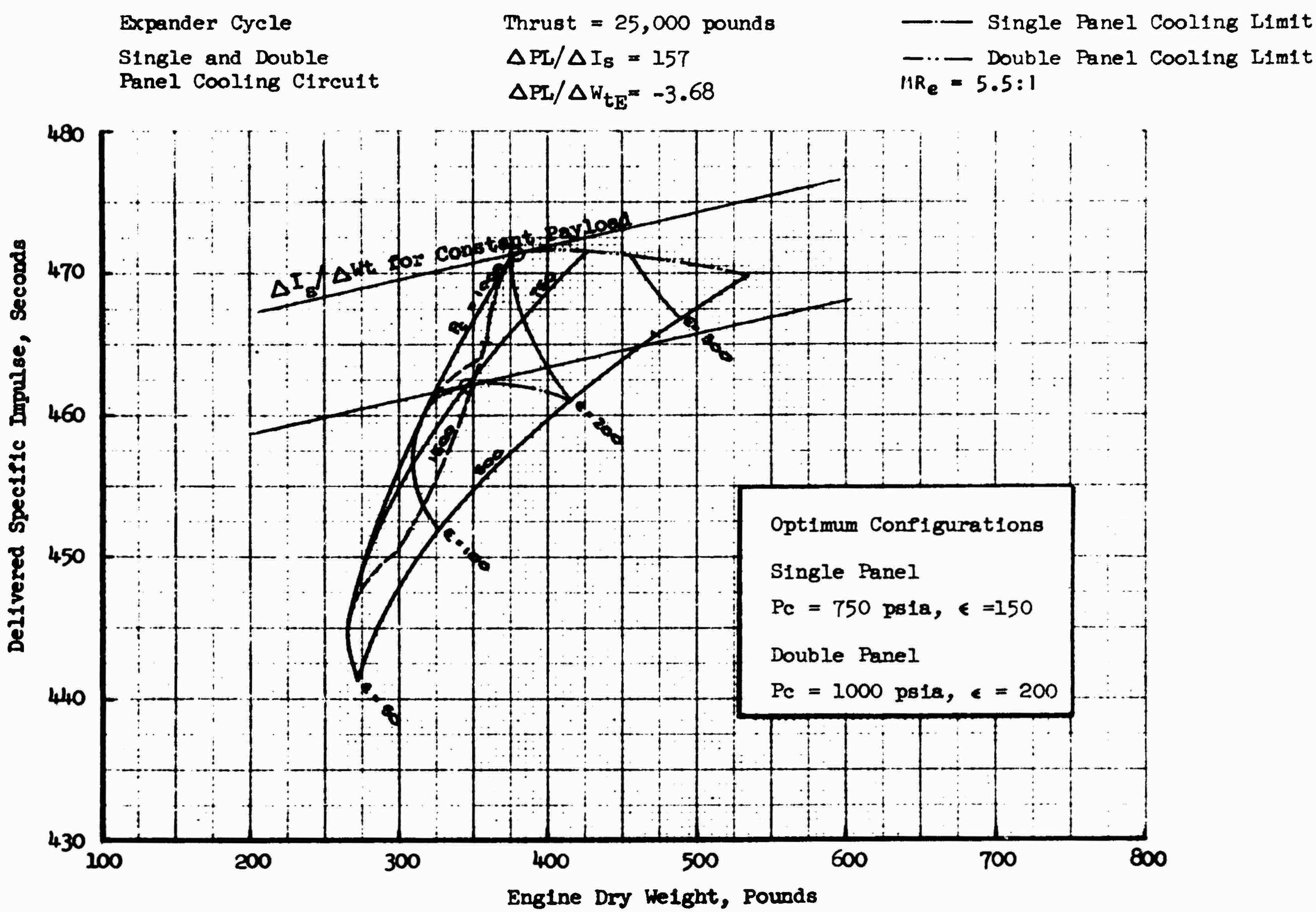


Figure 202. Aerospike Engine Optimization

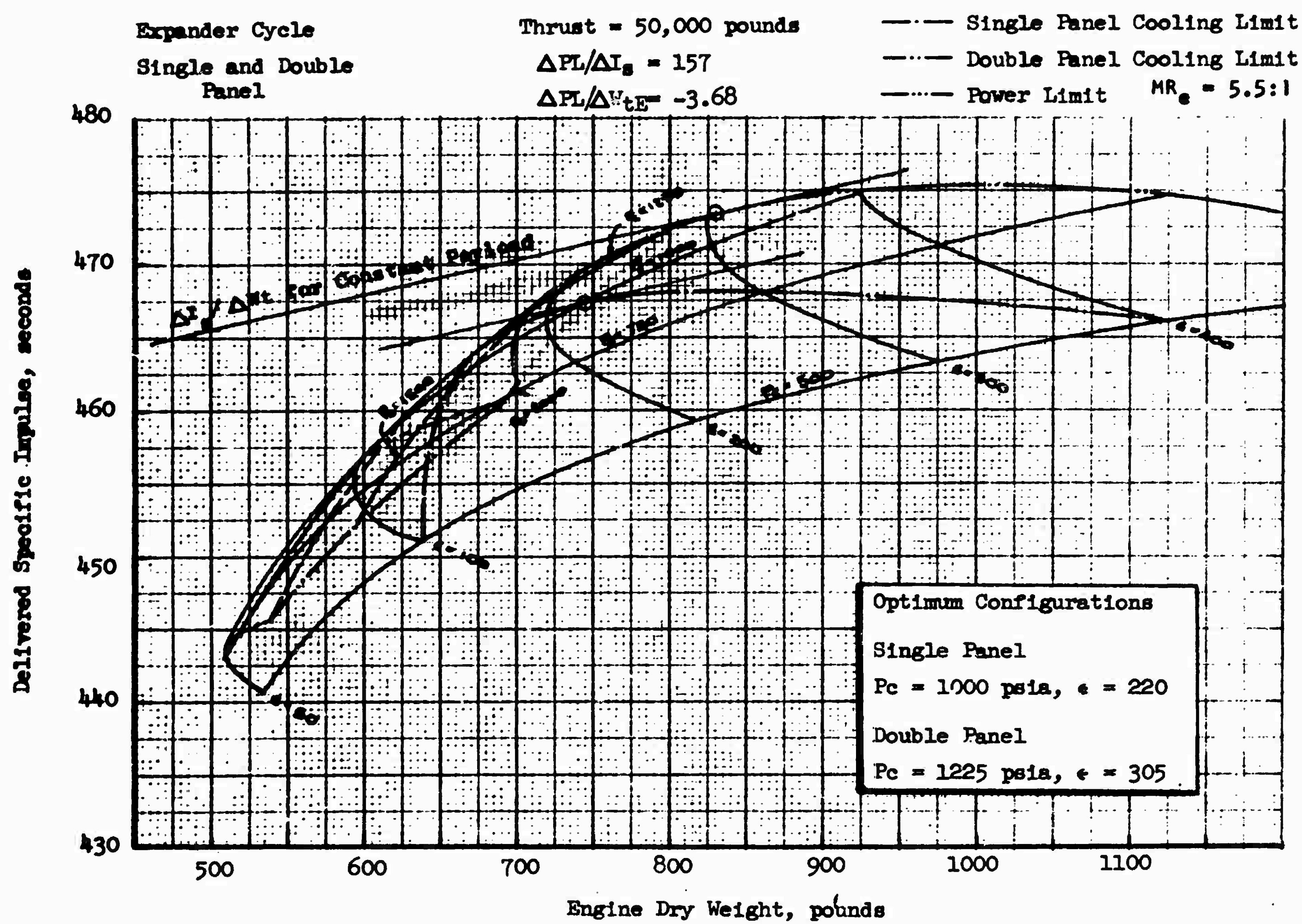


Figure 203. Aerospike Engine Optimization

TABLE 80. SELECTED O_2/H_2 AEROSPIKE ENGINE DESIGN POINTS

M.R = 5.5:1

Design Thrust Level	8,000 lb	15,000 lb	25,000 lb	50,000 lb
SINGLE-PANEL COOLING CIRCUIT				
T/P Cycle	Expander Topping	Expander Topping	Expander Topping	Expander Topping
P_c, psia	550	650	750	1000
Area Ratio	62	102	150	220
I_s, sec	449	456.6	462.1	467.3
Weight, lb	102	197	345	737
Length, in.	11	16	22.5	32.5
Diameter, in.	32	46	62	92
DOUBLE-PANEL COOLING CIRCUIT				
T/P Cycle	Expander Topping	Expander Topping	Expander Topping	Expander Topping
P_c, psia	500	800	1000	1225
Area Ratio	100	152	200	305
I_s, sec	452.0	466.6	471.1	473.4
Weight, lb	118	219	375	822
Length, in.	14	17.5	23	31.5
Diameter, in.	40	48	64	98

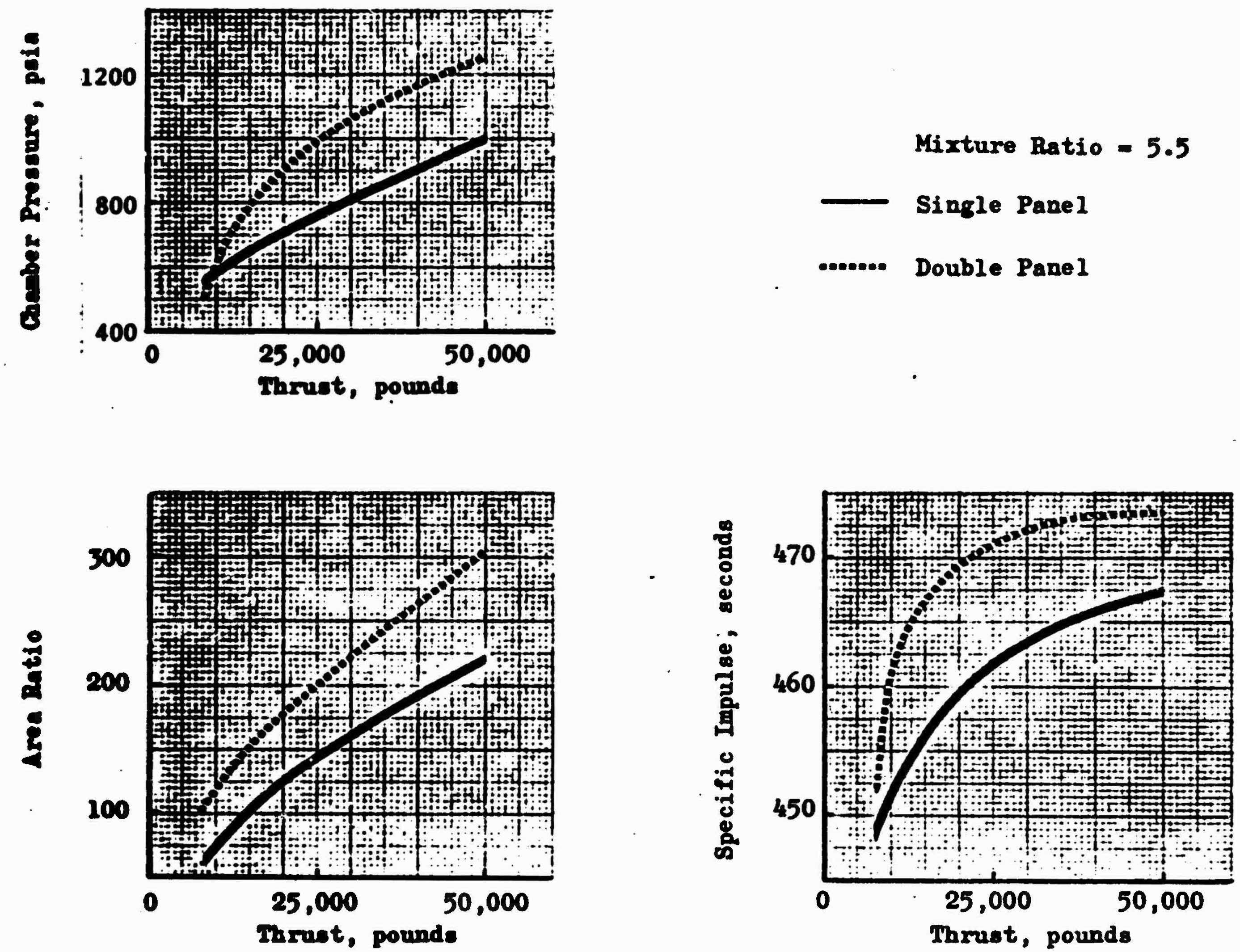


Figure 204. Selected Aerospike Engine Design Points

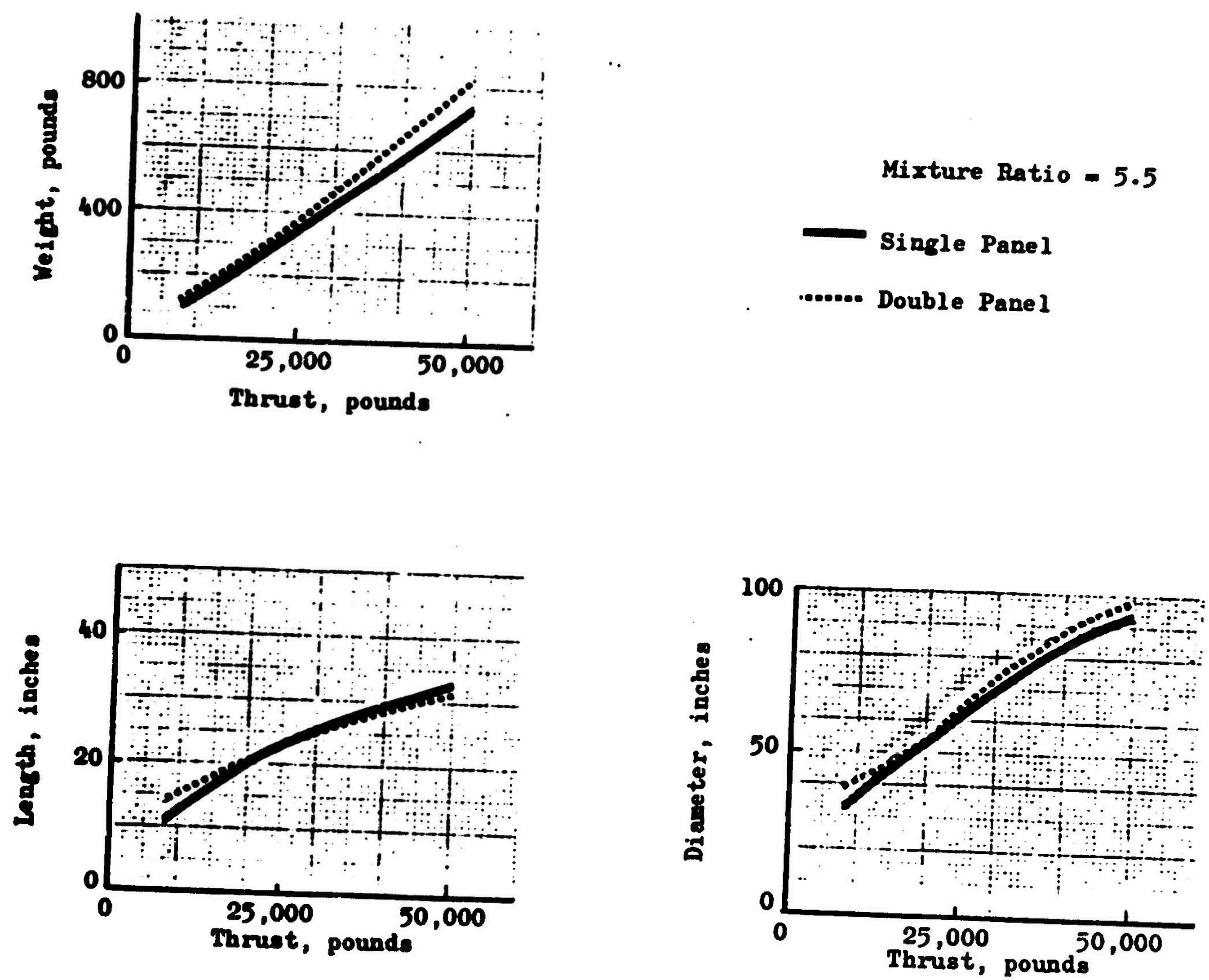


Figure 205. Selected Aerospike Engine Weights and Dimensions

ENGINE MIXTURE RATIO AND NPSH EFFECTS

The effect of variations in the design value of engine mixture ratio and available pump NPSH on engine performance, weight, dimensions, and turbopump operating conditions are presented in this section. Engine mixture ratios of 5, 6, and 7 and turbopump NPSH's of 16/60 (o/f), 9/37, and 2/15 feet were investigated. Low-pressure pumps also were considered. The available pump NPSH affects the design engine performance by constraining the pump rotational speeds. Low values of NPSH (low speeds) result in low turbopump efficiencies, reducing the maximum attainable chamber pressure for the expander cycle due to turbopump power limitations, and decreasing the delivered specific impulse for open cycles (gas generator and auxiliary heat exchanger) due to larger secondary flows.

OPEN CYCLES

The effect of NPSH on the specific impulse for open cycles is presented in Fig.206. The trends are applicable for the ranges of thrust, chamber pressure, area ratio, and mixture ratio indicated.

CLOSED CYCLES

Figure 207 presents the maximum chamber pressure for the expander cycle with single-panel cooling as a function of design thrust and available NPSH for an engine mixture ratio of 5.5 and a nozzle area ratio of 100. Also included are the effects of adding low-pressure pumps and varying the mixture ratio and area ratio (for the nominal NPSH values of 16/60). Approximately 80 percent of the increase in attainable chamber pressure due to the addition of low-pressure pumps is attributable to the fuel low-pressure pump. A representative line for double-panel cooling also is shown.

Aerospike engine system operating data were determined for parametric variations in the design engine mixture ratio and available turbopump design NPSH. Mixture ratios of 5, 6, and 7 and turbopump NPSH's of 16/60 (o/f, 9/37 and 2/15 feet were

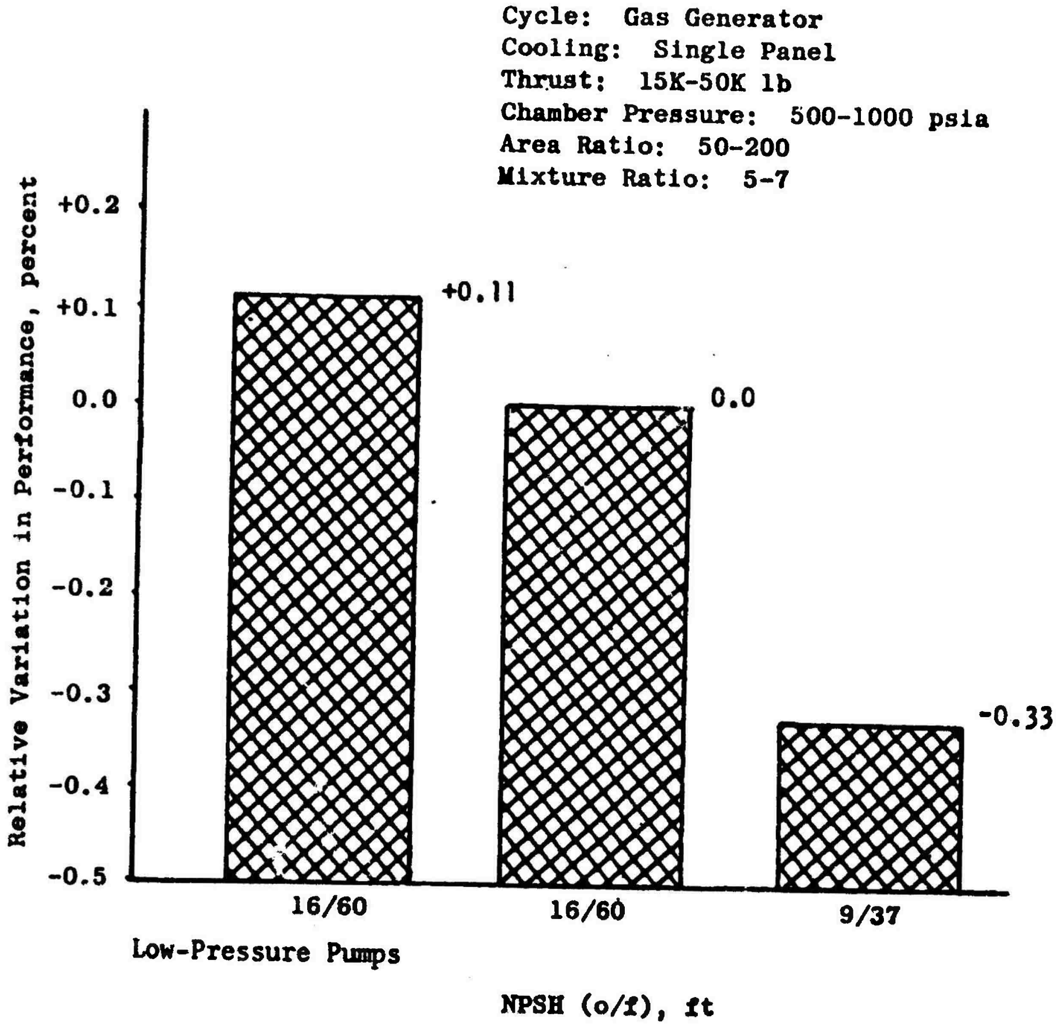


Figure 206. Effect of Available Pump NPSH on Delivered Vacuum Design Performance of Open Cycle Aerospike Engine

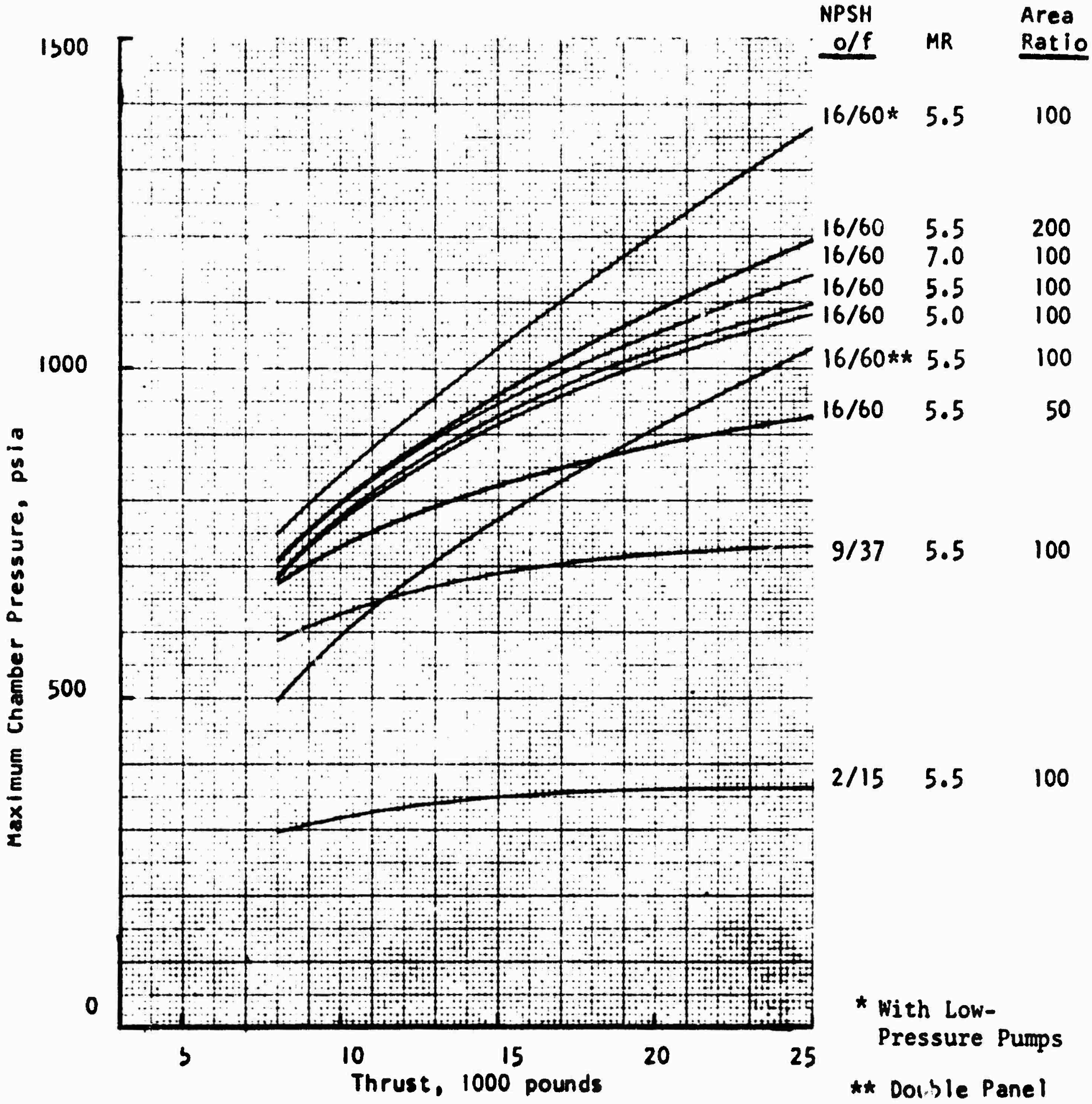


Figure 207. Effect of Available Pump NPSH on Maximum Chamber Pressure (Power Limit) for Single-Panel Aerospike Engine with Expander Topping Cycle

considered. The effects of utilizing low-pressure pumps for NPSH's of 16/60 feet also were investigated. Engine system data for this matrix of design conditions were determined for each of six previously selected optimum engine designs with thrust levels of 8000 pounds, 15,000 pounds, and 25,000 pounds and either single- or double-panel cooling circuits. These optimum engine systems had been selected based on a design mixture ratio of 5.5 and they all utilized expander topping turbopump drive cycles.

The engine system data are summarized in Tables 81 through 86. Data presented include chamber pressure, P_c (psia); nozzle area ratio, ε; delivered vacuum specific impulse, I_s (sec); dry weight, Wgt (lb); engine diameter, D_E (in.); engine length, L_E (in.); interface diameters, $D_{I,O}$, $D_{I,F}$ (in.); pump discharge pressures, $P_{D,OP}$, $P_{D,FP}$ (psia); and pump speeds, N_{OP}, N_{FP} (rpm). The values shown are percent variations from values for the baseline engines with a mixture ratio of 5.5 and NPSH's of 16/60 feet.

The chamber pressure and area ratio combinations were chosen to provide optimum engine performance based on a trade factor of 42.6 pounds of dry weight per second of specific impulse and subject to cooling and drive cycle power limits. Specific impulse and envelope dimensions were obtained by cross plotting parametric engine data presented in a separate report. Engine weights also were derived from the parametric data, but were corrected to represent more detailed turbopump designs than are incorporated in the parametric weight program. A method of driving the low-pressure pumps was not selected and, therefore, weights associated with this function were not included. Turbopump parameters were calculated with an optimization program which minimizes fuel pump discharge pressure for a specified minimum bypass flow (10 percent of the total fuel flow was used throughout this study). The turbopump optimizations were conducted subject to the design limits presented in Table 87.

TABLE 81. ENGINE SYSTEM DATA FOR DESIGN MIXTURE RATIO AND NPSH VARIATIONS, DOUBLE-PANEL, 8,000-POUNDS-THRUST

(values in percent variation from selected design point at MR = 5.5:1)

Thrust: 8,000 pounds Cycle: Expander Topping Cooling: Double Panel

MR	NPSH = 16/60 (o/f) with Low-Pressure Pumps	NPSH = 16/60 (o/f)	NPSH = 9/37 (o/f)	NPSH = 2/15 (o/f)
5	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +14 $P_{D,FP}$ = +4.7 I_s = +1.3 Wgt = +5.4 D_E = +6.1 L_E = +6.2 $D_{I,O}$ = +[illegible].1 N_{OP} = +7.9 $D_{I,F}$ = +31 N_{FP} = +2.1	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +14 $P_{D,FP}$ = +6.1 I_s = +1.3 Wgt = +5.4 D_E = +6.1 L_E = +6.2 $D_{I,O}$ = -2.7 N_{OP} = +9.7 $D_{I,F}$ = +3.4 N_{FP} = -0.1	Optimum design exceeds range of parametric data P_c = 400 psia	Optimum design exceeds range of parametric data P_c <400 psia
6	P_c = +11 $P_{D,OP}$ = +11 ϵ = -18 $P_{D,FP}$ = +15 I_s = -1.2 Wgt = -9.0 D_E = -12 L_E = -12 $D_{I,O}$ = +8.1 N_{OP} = +22 $D_{I,F}$ = +24 N_{FP} = +7.4	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -14 $P_{D,FP}$ = -6.1 I_s = -1.3 Wgt = -5.4 D_E = -6.1 L_E = -6.2 $D_{I,O}$ = +2.7 N_{OP} = -9.3 $D_{I,F}$ = +0.1 N_{FP} = +0.1	P_c = -11 $P_{D,OP}$ = -11 ϵ = -14 $P_{D,FP}$ = -18 I_s = -1.7 Wgt = -1.1 D_E = -0.6 L_E = +0.6 $D_{I,O}$ = +2.7 N_{OP} = -9.1 $D_{I,F}$ = +3.4 N_{FP} = -27	Optimum design exceeds range of parametric data P_c <400 psia
7	P_c = +11 $P_{D,OP}$ = +9.8 ϵ = -39 $P_{D,FP}$ = +6.1 I_s = -4.9 Wgt = -18 D_E = -22 L_E = -22 $D_{I,O}$ = +13.5 N_{OP} = +36 $D_{I,F}$ = +17 N_{FP} = +7.4	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -35 $P_{D,FP}$ = -14 I_s = -4.9 Wgt = -16 D_E = -16 L_E = -16 $D_{I,O}$ = +8.1 N_{OP} = -10 $D_{I,F}$ = -3.4 N_{FP} = 0.0	P_c = -11 $P_{D,OP}$ = -11 ϵ = -31 $P_{D,FP}$ = -20 I_s = -5.1 Wgt = -10 D_E = -9.4 L_E = -8.1 $D_{I,O}$ = +8.1 N_{OP} = -13 $D_{I,F}$ = +3.4 N_{FP} = -27	Optimum design exceeds range of parametric data P_c <400 psia

KEY

- P_c = chamber pressure, psia
- ϵ = area ratio
- I_s = specific impulse, seconds
- Wgt = engine weight, pounds
- D_E = engine diameter, inches
- L_E = engine length, inches
- D_I = inlet line inside diameter, inches
- N = pump speed, rpm
- P_d = pump discharge pressure, psia

SUBSCRIPTS:

- O = oxidizer
- F = fuel
- OP = oxidizer pump
- FP = fuel pump

TABLE 82. ENGINE SYSTEM DATA FOR DESIGN MIXTURE RATIO AND NPSH VARIATIONS, DOUBLE-PANEL, 15,000-POUNDS-THRUST

(values in percent variation from selected design point at MR = 5.5:1)

Thrust: 15,000 pounds Cycle: Expander Topping Cooling: Double Panel

MR	Parameter	NPSH = 16/60 (o/f) with Low-Pressure Pumps	NPSH = 16/60 (o/f)	NPSH = 9/37 (o/f)	NPSH = 2/15 (o/f)
5	P_c	= +20	= 0.0	= -13	Optimum design exceeds range of parametric data P_c <400 psia
	$P_{D,OP}$	= _23	= 0.0	= -14	
	ϵ	= -3.0	= 15	= +27	
	$P_{D,FP}$	= +51	= +5.1	= -17	
	I_s	= +1.1	= +1.1	= +0.9	
	Wgt	= -5.6	= 6.0	= +14	
	D_E	= -8.0	= +7.0	= +19	
	L_E	= -9.3	= +7.7	= +19	
	$D_{I,O}$	= +20	= -2.2	= +11	
	N_{OP}	= +66	= +7.3	= -26	
	$D_{I,F}$	= +32	= +5.3	= +16	
	N_{FP}	= +47	= -7.6	= -30	
6	P_c	= +13	= 0.0	= -13	Optimum design exceeds range of parametric data P_c <400 psia
	$P_{D,OP}$	= +15	= 0.0	= -14	
	ϵ	= -24	= -15	= -3.0	
	$P_{D,FP}$	= +13	= -5.1	= -21	
	I_s	= -1.1	= -1.1	= -1.1	
	Wgt	= -11	= -6.0	= +3.4	
	D_E	= -15	= -7.0	= +5.2	
	L_E	= -13	= -7.7	= +4.1	
	$D_{I,O}$	= +20	= +2.2	= +11	
	N_{OP}	= +53	= -7.3	= -23	
	$D_{I,F}$	= +26	= -5.3	= +11	
	N_{FP}	= +45	= +7.6	= -27	
7	P_c	= +13	= 0.0	= -13	Optimum design exceeds range of parametric data P_c <400 psia
	$P_{D,OP}$	= +15	= 0.0	= -14	
	ϵ	= -42	= -36	= -27	
	$P_{D,FP}$	= +7.5	= -9.6	= -23	
	I_s	= -4.4	= -4.4	= -4.5	
	Wgt	= -22	= -16	= -7.7	
	D_E	=-25	= -19	= -7.0	
	L_E	= -26	= -19	= -9.3	
	$D_{I,O}$	= +24	= +2.2	= +11	
	N_{OP}	= +49	= +1.0	= -25	
	$D_{I,F}$	= +21	= -5.3	= +5.3	
	N_{FP}	= +47	= +8.8	= -23	

KEY

P_c = chamber pressure, psia
ϵ = area ratio
I_s = specific impulse, seconds
Wgt = engine weight, pounds
D_E = engine diameter, inches
L_E = engine length, inches
D_I = inlet line inside diameter, inches
N = pump speed, rpm
P_d = pump discharge pressure, psia

SUBSCRIPTS:
O = oxidizer
F = fuel
OP = oxidizer pump
FP = fuel pump

TABLE 83. ENGINE SYSTEM DATA FOR DESIGN MIXTURE RATIO AND NPSH VARIATIONS, DOUBLE-PANEL, 25,000-POUNDS-THRUST

(values in percent variation from selected design point at MR = 5.5:1)

Thrust: 25,000 pounds Cycle: Expander Topping Cooling: Double Panel

MR	NPSH = 16/60 (o/f) with Low-Pressure Pumps	NPSH = 16/60 (o/f)	NPSH = 9/37 (o/f)	NPSH = 2/15 (o/f)
5	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +15 $P_{D,FP}$ = -3.4 I_s = +0.9 Wgt = +1.9 D_E = +7.0 L_E = +7.1 $D_{I,O}$ = +26 N_{OP} = +70 $D_{I,F}$ = +35 N_{FP} = +16	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +15 $P_{D,FP}$ = +2.4 I_s = +0.9 Wgt = +7.0 D_E = +7.0 L_E = +7.1 $D_{I,O}$ = 0.0 N_{OP} = -12 $D_{I,F}$ = +2.0 N_{FP} = -3.5	P_c = -15 $P_{D,OP}$ = -17 ϵ = +40 $P_{D,FP}$ = -19 I_s = +0.9 Wgt = +19 D_E = +27 L_E = +30 $D_{I,O}$ = +15 N_{OP} = -24 $D_{I,F}$ = +18 N_{FP} = -33	Optimum design exceeds range of parametric data $P_c \approx 450$ ϵ > 400
6	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -15 $P_{D,FP}$ = -7.4 I_s = -0.9 Wgt = -9.9 D_E = -7.0 L_E = -7.1 $D_{I,O}$ = +30 N_{OP} = +60 $D_{I,F}$ = +27 N_{FP} = +24	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -15 $P_{D,FP}$ = -2.4 I_s = -0.9 Wgt = -7.0 D_E = -7.0 L_E = -7.1 $D_{I,O}$ = 0.0 N_{OP} = +12 $D_{I,F}$ = -2.0 N_{FP} = +3.5	P_c = -20 $P_{D,OP}$ = -22 ϵ = +10 $P_{D,FP}$ = -28 I_s = -1.0 Wgt = +6.1 D_E = +16 L_E = +17 $D_{I,O}$ = +19 N_{OP} = -39 $D_{I,F}$ = +10 N_{FP} = -28	P_c = -55 $P_{D,OP}$ = -59 ϵ = +60 $P_{D,FP}$ = -57 I_s = -1.8 Wgt = +55 D_E = +86 L_E = +84 $D_{I,O}$ = +70 N_{OP} = -76 $D_{I,F}$ = +39 N_{FP} = -65
7	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -35 $P_{D,FP}$ = -9.8 I_s = -4.0 Wgt = -19 D_E = -17 L_E = -18 $D_{I,O}$ = +33 N_{OP} = +62 $D_{I,F}$ = +18 N_{FP} = +16	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -35 $P_{D,FP}$ = -0.3 I_s = -4.0 Wgt = -15 D_E = -17 L_E = -18 $D_{I,O}$ = +3.7 N_{OP} = +11 $D_{I,F}$ = -2.0 N_{FP} = -6.8	P_c = -20 $P_{D,OP}$ = -22 ϵ = -15 $P_{D,FP}$ = -22 I_s = -4.0 Wgt = -2.1 D_E = +3.1 L_E = +2.4 $D_{I,O}$ = +19 N_{OP} = -28 $D_{I,F}$ = +6.1 N_{FP} = -32	P_c = -55 $P_{D,OP}$ = -59 ϵ = +25 $P_{D,FP}$ = -59 I_s = -4.9 Wgt = +39 D_E = +64 L_E = +61 $D_{I,O}$ = +74 N_{OP} = -77 $D_{I,F}$ = +31 N_{FP} = -63

KEY

P_c = chamber pressure, psia
ϵ = area ratio
I_s = specific impulse, seconds
Wgt = engine weight, pounds
D_E = engine diameter, inches
L_E = engine length, inches
D_I = inlet line inside diameter, inches
N = pump speed, rpm
P_d = pump discharge pressure, psia

SUBSCRIPTS:

O = oxidizer
F = fuel
OP = oxidizer pump
FP = fuel pump

TABLE 84. ENGINE SYSTEM DATA FOR DESIGN MIXTURE RATIO AND NPSH VARIATIONS, SINGLE-PANEL, 8,000-POUNDS-THRUST

(values in percent variation from selected design point at MR = 5.5:1)

Thrust: 8,000 pounds   Cycle: Expander Topping   Cooling: Single Panel

MR	NPSH = 16/60 (o/f) with Low-Pressure Pumps	NPSH = 16/60 (o/f)	NPSH = 9/37 (o/f)	NPSH = 2/16 (o/f)
5	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +17 $P_{D,OP}$ = +3.3 I_s = +1.4 Wgt = +3.7 D_E = +5.6 L_E = +5.4 $D_{I,O}$ = +18 N_{OP} = +17 $D_{I,F}$ = +31 N_{FP} = +1.0	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +17 $P_{D,FP}$ = +4.2 I_s = +1.4 Wgt = +1.7 D_E = +5.6 L_E = +5.4 $D_{I,O}$ = 0.0 N_{OP} = +1.3 $D_{I,F}$ = +3.4 N_{FP} = -4.2	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = +17 $P_{D,FP}$ = +19 I_s = +1.4 Wgt = +13.1 D_E = +5.6 L_E = +5.4 $D_{I,O}$ = +12 N_{OP} = -24 $D_{I,F}$ = +17 N_{FP} = -31	Optimum design exceeds range of parametric data P_c <400 psia
6	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -17 $P_{D,FP}$ = -5.1 I_s = -1.4 Wgt = -5.6 D_E = -5.6 L_E = -5.4 $D_{I,O}$ = +18 N_{OP} = +19 $D_{I,F}$ = +24 N_{FP} = +9.7	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -17 $P_{D,FP}$ = -4.2 I_s = -1.4 Wgt = -4.7 D_E = -5.6 L_E = -5.4 $D_{I,O}$ = 0.0 N_{OP} = -1.3 $D_{I,F}$ = 0.0 N_{FP} = +4.2	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -17 $P_{D,FP}$ = +10 I_s = -1.4 Wgt = +3.7 D_E = -5.6 L_E = -5.4 $D_{I,O}$ = +12 N_{OP} = -27 $D_{I,F}$ = +10 N_{FP} = -27	Optimum design exceeds range of parametric data P_c <400 psia
7	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -33 $P_{D,FP}$ = -9.5 I_s = -4.6 Wgt = -9.3 D_E = -12 L_E = -12 $D_{I,O}$ = +24 N_{OP} = +15 $D_{I,F}$ = +17 N_{FP} = +9.5	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -33 $P_{D,FP}$ = -5.5 I_s = -4.6 Wgt = -4.7 D_E = -12 L_E = -12 $D_{I,O}$ = +5.9 N_{OP} = -4.5 $D_{I,F}$ = -3.4 N_{FP} = -18	P_c = 0.0 $P_{D,OP}$ = 0.0 ϵ = -33 $P_{D,FP}$ = -0.7 I_s = -4.6 Wgt = -1.9 D_E = -12 L_E = -12 $D_{I,O}$ = +12 N_{OP} = -21 $D_{I,F}$ = +3.4 N_{FP} = -21	Optimum design exceeds range of parametric data P_c <400 psia

KEY

- P_c = chamber pressure, psia
- ϵ = area ratio
- I_s = specific impulse, seconds
- Wgt = engine weight, pounds
- D_E = engine diameter, inches
- L_E = engine length, inches
- D_I = inlet line inside diameter, inches
- N = pump speed, rpm
- P_d = pump discharge pressure, psia

SUBSCRIPTS:

- O = oxidizer
- F = fuel
- OP = oxidizer pump
- FP = fuel pump

TABLE 85. ENGINE SYSTEM DATA FOR DESIGN MIXTURE RATIO AND NPSH VARIATIONS, SINGLE-PANEL, 15,000-POUNDS-THRUST

(values in percent variation from selected design point at MR = 5.5:1)

Thrust: 15,000 pounds   Cycle: Expander Topping   Cooling: Single Panel

MR	Parameter	NPSH = 16/60 (o/f) with Low-Pressure Pumps	NPSH = 16/60 (o/f)	NPSH = 9/37 (o/f)	NPSH = 2/15 (o/f)
5	P_c	0.0	0.0	0.0	-39
	$P_{D,OP}$	0.0	0.0	0.0	-38
	ϵ	+15	+15	+15	+50
	$P_{D,FP}$	+0.3	+1.5	+4.5	=36
	I_s	+1.2	+1.2	+1.2	+0.7
	Wgt	+2.6	+4.1	+8.7	+47
	D_E	+6.6	+6.6	+6.6	+54
	L_E	+6.8	+6.8	+6.8	+51
	$D_{I,O}$	+15	=2.1	+6.4	+53
	N_{OP}	+30	+6.5	-18	-71
	$D_{I,F}$	+33	+2.6	+18	+44
	N_{FP}	+7.0	-2.6	-29	-64
6	P_c	0.0	0.0	0.0	-38
	$P_{D,OP}$	0.0	0.0	0.0	-38
	ϵ	-15	-15	-15	+15
	$P_{D,FP}$	-3.9	-1.5	+3.3	-34
	I_s	-1.2	-1.2	-1.2	-1.5
	Wgt	-6.7	-4.1	+0.5	+34
	D_E	-6.6	-6.6	-6.6	+36
	L_E	-6.8	-6.8	-6.8	+33
	$D_{I,O}$	+19	+2.1	+6.4	+53
	N_{OP}	+34	-6.5	-12	-71
	$D_{I,F}$	+23	-2.6	+13	+38
	N_{FP}	+16	+2.0	-35	-62
7	P_c	0.0	0.0	0.0	Optimum design exceeds range of parametric data P_c < 400 psia
	$P_{D,OP}$	0.0	0.0	0.0	
	ϵ	-35	-35	-35	
	$P_{D,FP}$	-6.8	-2.7	+2.1	
	I_s	-4.5	-4.5	-4.5	
	Wgt	-14	-11	-7.2	
	D_E	-16	-16	-16	
	L_E	-17	-17	-17	
	$D_{I,O}$	+19	+2.1	+6.4	
	N_{OP}	+27	+11	-12	
	$D_{I,F}$	+18	-2.6	+7.7	
	N_{FP}	+30	-17	-32	

KEY

P_c = chamber pressure, psia
ϵ = area ratio
I_s = specific impulse, seconds
Wgt = engine weight, pounds
D_E = engine diameter, inches
L_E = engine length, inches
D_I = inlet line inside diameter, inches
N = pump speed, rpm
P_d = pump discharge pressure, psia

SUBSCRIPTS:
O = oxidizer
F = fuel
OP = oxidizer pump
FP = fuel pump

TABLE 86. ENGINE SYSTEM DATA FOR DESIGN MIXTURE RATIO AND NPSH VARIATIONS, SINGLE-PANEL, 25,000-POUNDS-THRUST

(values in percent variation from selected design point at MR = 5.5:1)

Thrust: 25,000 pounds Cycle: Expander Topping Cooling: Single Panel

MR	NPSH = 16/60 (o/f) with Low-Pressure Pumps	NPSH = 16/60 (o/f)	NPSH = 9/37 (o/f)	NPSH = 2/15 (o/f)
5	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = -40 $P_{D,OP}$ = -40
	ϵ = +13 $P_{D,FP}$ = -1.3	ϵ = +13 $P_{D,FP}$ = +1.3	ϵ = +13 $P_{D,FP}$ = +7.0	ϵ = +60 $P_{D,FP}$ = -27
	I_s = +1.0	I_s = +1.0	I_s = +1.0	I_s = +0.5
	Wgt = +1.7	Wgt = +5.7	Wgt = +6.9	Wgt = +53
	D_E = +5.9	D_E = +5.9	D_E = +5.9	D_E = +62
	L_E = +6.8	L_E = +6.8	L_E = +6.8	L_E = +60
	$D_{I,O}$ = +13 N_{OP} = +26	$D_{I,O}$ = 0.0 N_{OP} = 0.0	$D_{I,O}$ = +3.2 N_{OP} = -28	$D_{I,O}$ = +45 N_{OP} = -75
	$D_{I,F}$ = +25 N_{FP} = -1.4	$D_{I,F}$ = +1.9 N_{FP} = -0.1	$D_{I,F}$ = +13 N_{FP} = -25	$D_{I,F}$ = +43 N_{FP} = -63
6	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = -40 $P_{D,OP}$ = -40
	ϵ = -13 $P_{D,FP}$ = -3.8	ϵ = -13 $P_{D,FP}$ = -1.3	ϵ = -13 $P_{D,FP}$ = +5.1	ϵ = +20 $P_{D,FP}$ = -28
	I_s = -1.0	I_s = -1.0	I_s = -1.0	I_s = -1.6
	Wgt = -9.0	Wgt = -5.7	Wgt = +0.9	Wgt = +37
	D_E = -5.9	D_E = -5.9	D_E = -5.9	D_E = +40
	L_E = -6.8	L_E = -6.8	L_E = -6.8	L_E = +38
	$D_{I,O}$ = +13 N_{OP} = +31	$D_{I,O}$ = 0.0 N_{OP} = -3.7	$D_{I,O}$ = +3.2 N_{OP} = -25	$D_{I,O}$ = +48 N_{OP} = -76
	$D_{I,F}$ = +17 N_{FP} = +27	$D_{I,F}$ = -1.9 N_{FP} = +0.1	$D_{I,F}$ = +13 N_{FP} = -34	$D_{I,F}$ = +32 N_{FP} = -59
7	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = 0.0 $P_{D,OP}$ = 0.0	P_c = -40 $P_{D,OP}$ = -40
	ϵ = -33 $P_{D,FP}$ = -4.5	ϵ = -33 $P_{D,FP}$ = -2.5	ϵ = -33 $P_{D,FP}$ = +3.2	ϵ = -6.7 $P_{D,FP}$ = -31
	I_s = -4.1	I_s = -4.1	I_s = -4.1	I_s = -4.6
	Wgt = -17	Wgt = -11	Wgt = -6.3	Wgt = +26
	D_E = -16	D_E = -16	D_E = -16	D_E = +25
	L_E = -17	L_E = -17	L_E = -17	L_E = +22
	$D_{I,O}$ = +16 N_{OP} = +66	$D_{I,O}$ = +3.2 N_{OP} = -1.1	$D_{I,O}$ = +9.7 N_{OP} = -33	$D_{I,O}$ = +55 N_{OP} = -77
	$D_{I,F}$ = +9.4 N_{FP} = +42	$D_{I,F}$ = -1.9 N_{FP} = -12	$D_{I,F}$ = +1.9 N_{FP} = -23	$D_{I,F}$ = +25 N_{FP} = -54

KEY

P_c = chamber pressure, psia
ϵ = area ratio
I_s = specific impulse, seconds
Wgt = engine weight, pounds
D_E = engine diameter, inches
L_E = engine length, inches
D_I = inlet line inside diameter, inches
N = pump speed, rpm
P_d = pump discharge pressure, psia

SUBSCRIPTS:
O = oxidizer
F = fuel
OP = oxidizer pump
FP = fuel pump

TABLE 87. TURBOMACHINERY DESIGN LIMITS

Pump	
DN, mm-rpm	$\leq 2 \times 10^6$
Tip Width, inch	≥ 0.030
Hub Diameter/Tip Diameter	≥ 0.8
Specific Speed	$\geq 400, \leq 2000$
Turbines	
Tip Speed, ft/sec	≤ 1700
Blade Height, inch	≥ 0.150
Stress Parameter AAN^2, in^2 -rpm^2	$\leq 50 \times 10^9$
Pitch Diameter, inch	≥ 2
Hub Diameter/Tip Diameter	≤ 0.9
Pitch Diameter/Pump Impeller Diameter	3

Most of the effects of varying the design mixture ratio and/or NPSH are evident in Table 84. In most instances, low-pressure pumps do not offer any increase in engine performance when compared with the corresponding case with NPSH's of 16/60 feet because the optimum is controlled by cooling limits rather than the drive cycle power limits. Engines with design thrusts of 8000 pounds and double-panel cooling are exceptions and small increases can be achieved with low-pressure pumps. As the pump NPSH's are decreased, a point is reached where insufficient power is available to drive the turbopumps due to low efficiencies at the reduced speeds. At this point, it is necessary to reduce the chamber pressure. Part of the resulting loss in specific impulse can be regained by increasing the area ratio to the new cooling limit, but the weight and envelope dimensions also increase. Decreasing the NPSH's also results in increased pump inlet diameters.

The most significant effect of changing the design mixture ratio is the large variation in specific impulse. Specific impulse decreases with increasing mixture ratio due to theoretical performance considerations and, also, as a result of decreasing the area ratio to meet cooling limitations with the reduced hydrogen

flow. The smaller area ratio at a higher mixture ratio does result in a lighter weight and smaller envelope dimensions; however, these advantages are comparatively minor.

A few of the combinations of mixture ratio and NPSH's resulted in engine designs which exceeded the ranges of accurate parametric data and, therefore, were omitted. Extrapolations of reported data indicate these designs are unattractive with respect to performance, weight, and size.

REFERENCES

1. R-8786, O_2H_2 Advanced Maneuvering Propulsion Technology Program, Aerospike Engine Configuration Design and Analysis, Rocketdyne, a division of North American Rockwell Corporation, Canoga Park, California, 31 May 1971.

2. MCR-70-150, Final Report, Study of Space Station Propulsion System Resupply and Repair, Martin Marietta, Contract NAS8-25067, June 1970.

3. R-7888P, Study of Space Station Propulsion System Resupply and Repair, Rocketdyne, a division of North American Rockwell Corporation, Canoga Park, California, 1969.

4. NR Documents, SD68-304, -1, -2, -3, -4, Extravehicular Engineering Activities (EVA) Program Requirements (NAS8-18128 contract)

5. R-8083, Thermodynamic Improvements in Liquid Hydrogen Turbopumps, Rocketdyne, a division of North American Rockwell Corporation, Canoga Park, California, December 1969.

6. R-8283, Design of Inducers for Two-Phase Operation, Rocketdyne, a division of North American Rockwell Corporation, Canoga Park, California, July 1970.

7. Richmond, J., "Spectrographic Analysis of Detonation Wave Structures," Detonation and Two-Phase Flow, Academic Press, New York, N.Y., 1961.

APPENDIX A

CONTROL SYSTEM CONSIDERATIONS IN CYCLE SELECTION FOR 25,000-POUND-THRUST ENGINE

Four candidate engine configurations for the 25K O_2/H_2 aerospike AMPT engine were examined to determine control system considerations which could influence the cycle selection. Expander topping and gas generator cycles with parallel and geared turbopumps were studied. An operationally suitable control system for each of the candidate configurations was selected utilizing experience gained from previous analytical studies. Comparisons between characteristics associated with the four configurations are discussed and were integrated with other investigations in the final cycle selection.

DISCUSSION

The engine operating requirements which guided this study were throttling capability of 5:1, mixture ratio excursions between 5:1 and 6:1, and multiple restarts with coast times between 10 minutes and 14 days.

Control Method Selection

Each of the four candidate engine configurations was examined to select an operationally suitable control system.

Expander Topping Cycle. A flow schematic for the expander topping cycle is shown in Fig. A-1.

Oxidizer flow is ducted from the pump directly to the thrust chamber. Fuel flows from the pump to the nozzle and is used as the regenerative coolant. The heated hydrogen is then used to power the turbine drive system. The turbine exhaust is ducted to the thrust chamber injector and nozzle base. A control valve is indicated to regulate the base flow and provide optimum performance over the full

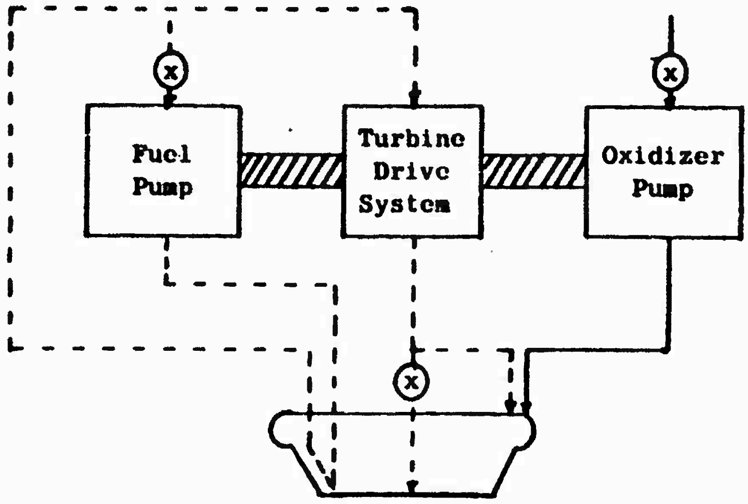


Figure A-1. Flow Schematic for Expander Topping Cycle

operating range of the engine. However, a fixed orifice could be used with little or no performance degradation. Propellant prevalves were assumed for isolation of the engine system from the vehicle.

Four possible methods of providing throttling capability for this cycle are: (1) liquid line propellant valves, (2) turbine system inlet valve(s), (3) turbine system exhaust valve(s) and (4) a turbine system bypass valve. Liquid line control valves normally result in the fastest throttling response, but increase pump discharge pressures. The increased pressures increase system weights and decrease stall margins at throttled conditions. Turbine system inlet and exhaust control valves also increase pump discharge pressures. In the absence of specific throttling rate requirements, a turbine bypass valve is attractive for thrust control because it does not increase pump discharge pressures.

In an earlier engine system study, dynamic model results showed that an engine operating on this cycle with a bypass valve for thrust control was capable of 10:1 throttling in 1.5 seconds. A disadvantage of turbine bypass control is that flow control becomes increasingly insensitive to a given valve area change during deep throttling.

Mixture ratio excursions require an additional control valve. This control element could be positioned in either the fuel or oxidizer propellant system. For the geared pump configuration, it is advantageous to position the valve in the oxidizer line so that the oxygen can be dropped to the pump at the same time the hydrogen prevalve is opened. This provides adequate lubrication to the oxidizer pump in the event of turbine rotation during fuel-lead chilldown and prevents oxygen from entering the thrust chamber where high mixture ratios could occur before fuel flow is established.

With a parallel turbopump arrangement, an oxidizer turbine inlet valve may be more attractive for mixture ratio control than a valve in the oxidizer propellant line. The disadvantages of liquid line control are the increases in pump discharge pressure and system weights, and the decrease in stall margin at throttled conditions. While these effects also exist with a turbine inlet valve, they are minimized by altering the turbine flow split at design.

Gas Generator Cycle. A flow schematic for the gas generator cycle is shown in Fig.A-2. Oxidizer flow is ducted from the pump to the thrust chamber and the gas generator. Fuel flows from the pump to the nozzle and is used as the regenerative coolant. The major portion of the heated hydrogen is ducted to the thrust chamber and the remainder to the gas generator. The low mixture ratio gas generator exhaust powers the turbopumps and is dumped in the nozzle base.

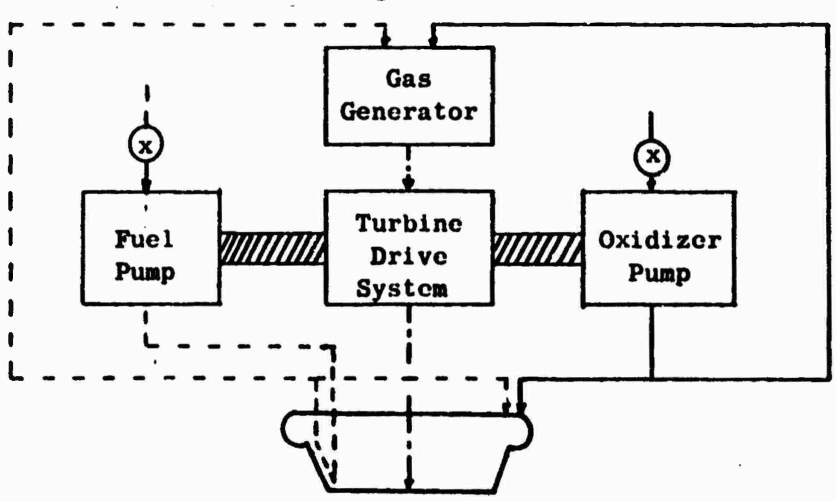


Figure A-2. Flow Schematic for Gas Generator Cycle

It should be pointed out that this particular gas generator cycle was chosen because of gas generator injector throttling requirements, which are relaxed by using heated hydrogen. However, the cooling jacket resistance to gas generator fuel flow during start may be critical.

A minimum of three control valves are required for control of thrust, gas generator mixture ratio, and main chamber mixture ratio. To minimize pump discharge pressures, thrust control elements should be positioned in the turbine drive circuit, not in the main propellant lines. Hot-gas valves can be avoided by using valves in each of the gas generator propellant lines. They also can serve as a gas generator mixture ratio controller. The disadvantage of using these valves is the reduction in available pressure for the turbine drive system. This increases the required turbine flow and could penalize specific impulse. Using heated hydrogen gas generator flow, instead of liquid fuel, has the same effect.

A valve in the main chamber oxidizer line is preferred for mixture ratio control of a geared-pump arrangement. Additional valve pressure losses due to mixture ratio control at throttled conditions can be tolerated better by the oxidizer pump because oxidizer system component pressure losses decrease more rapidly than fuel system losses as thrust is decreased. Also, because the main chamber mixture ratio tends to be high during start, a need to restrict oxidizer flow is implied because fuel system resistances are mostly due to flow in the coolant passages, which cannot be reduced to increase fuel flow.

This method of mixture ratio control also could be used with a parallel turbopump configuration; however, it is not as effective as a hot-gas oxidizer turbine inlet valve during start. A hot-gas valve would starve the oxidizer turbine and divert the flow to the fuel turbine, which tends to speed up the start; while a main chamber oxidizer valve restricts the oxidizer flow, increases the required pump power, and slows the start transient. A two-position main chamber oxidizer valve is still required, though, to ensure maximum gas generator flow during start.

A summary description of the control systems selected for the candidate cycles is shown in Table A-1.

TABLE A-1. SELECTED CONTROL SYSTEMS

Cycle	Expander Topping	Expander Topping	Gas Generator	Gas Generator
Turbopump Configuration	Geared	Parallel	Geared	Parallel
Valve Locations				
Thrust	Turbine Bypass	Turbine Bypass	GG Inlets(2)	GG Inlets(2)
Engine MR	O_2 Main Line	O_2 Turbine Inlet	TC O_2 Inlet	O_2 Turbine Inlet
GG MR			GG O_2 Inlet	GG O_2 Inlet
Engine System Isolation	Pump Inlets (2)	Pump Inlets (2)	Pump Inlets (2)	Pump Inlets (2)
Start Control				Thrust Chamber O_2 Inlet
Number of Valves				
Liquid	3	2	4	4
Gaseous H_2	1	2	1	1
Hot Gas				1

Feedback Point Selection

Parameters to be monitored as indicators of controlled variables will be discussed briefly. Chamber pressure is a practical indicator of engine thrust and can be measured reliably. An actual thrust measurement is dependent on the engine/vehicle interface and, therefore, not suitable for providing feedback for control purposes.

Main propellant flowmeters with temperature and pressure correction, if required, are adequate for determining thrust chamber mixture ratio. Hot-gas measuring devices for this purpose are beyond the state of the art. However, the space shuttle main engine is to utilize temperature measurements for precombustor

mixture ratio control, and, therefore, they should be considered for gas generator mixture ratio control. Problems which must be overcome are slow response and the use of a local measurement to indicate gross conditions.

Other devices are likely to be required to provide information on pump discharge propellant qualities during childown and oxidizer injector manifold priming.

Start Method

Two basic start methods are available for the engine cycles being studied. They utilize either tank-head or auxiliary power as an initial turbine energy source. Auxiliary power includes pressurized liquid storage bottles, gas spin bottles, and solid propellant spinners, none of which are applicable for high-flow turbines such as used in expander topping cycles. Problems associated with auxiliary power sources include refill for restart capability and added weight. Tank-head power usually results in a relatively long start time and is critically dependent on available tank pressures and thermal conditioning requirements. Also, control of the small flows under tank head with mainstage valves can be a source of problems.

In the absence of specific start time and preconditioning requirements, a tank-head start is recommended for both the cycles under investigation. Auxiliary power sources were eliminated on the basis of complexity associated with multiple restart capability and system weight. It should be noted that, while a tank-head start is conceptually simple, considerable effort and complex sensors may be required to develop a common start sequence which is adequate for the full range of engine initial conditions.

APPENDIX B

ENGINE SYSTEM COMPONENTS FAILURE MODE AND EFFECT ANALYSIS (FMEA)

INTRODUCTION

A failure mode and effect analysis (FMEA) was performed for all the major components of the O_2/H_2 AMPT aerospike engine. The results of these analyses are presented herein.

The purpose of the analysis is to investigate the adequacy of a design to meet its requirements by an assessment of the consequences and potential seriousness of the possible occurrence of each of the failure modes on the successful operation of the engine and vehicle and/or on the successful completion of the planned mission.

ENGINE SYSTEM DESCRIPTION

The aerospike thrust chamber assembly is regeneratively cooled of either single-panel or double-panel construction. The combustion chamber is composed of 24 cast segments which are ignited by a combustion wave igniter system. The combustor segments are bolted between two structural rings. The nozzle is of tubular construction.

The system employs centrifugal pumps for both propellants. Each pump is directly driven by a low-pressure-ratio turbine. The expander topping cycle is used so heated hydrogen from the thrust chamber cooling jacket drives the turbines in a parallel flow arrangement before it is injected into the main combustion chamber.

The control system consists of two main propellant valves and two turbine flow control valves. The main propellant valves are located upstream of the turbopumps

to provide positive propellant shutoff for extended coast capability in space. The turbine control valves provide both thrust and mixture ratio control. The main propellant valves are pneumatically actuated while the turbine control valves are electrically actuated. A pneumatic system using stage-supplied helium is provided for oxidizer system purge, oxidizer pump seal purge, and main propellant valve actuation.

The engine system schematic (Fig.B-1) shows the components and component interconnections of the aerospike engine system.

ANALYSIS PROCEDURE

The FMEA consists of five basic operations:

1. Identification of components
2. Description of component function
3. Identification of possible failure modes
4. Identification of possible causes for each failure mode
5. Prediction of possible failure effect on engine, vehicle, and mission coordinated, as applicable, by time of occurrence during mission sequence

In this analysis, structural failures and double failures were not considered. Stage-supplied propellants and pneumatic helium were assumed to meet the requirements for successful engine operation.

The failure effects were evaluated for five mission phases:

1. Preflight checkout
2. Start sequence
3. Mainstage
4. Cutoff
5. Orbital coast

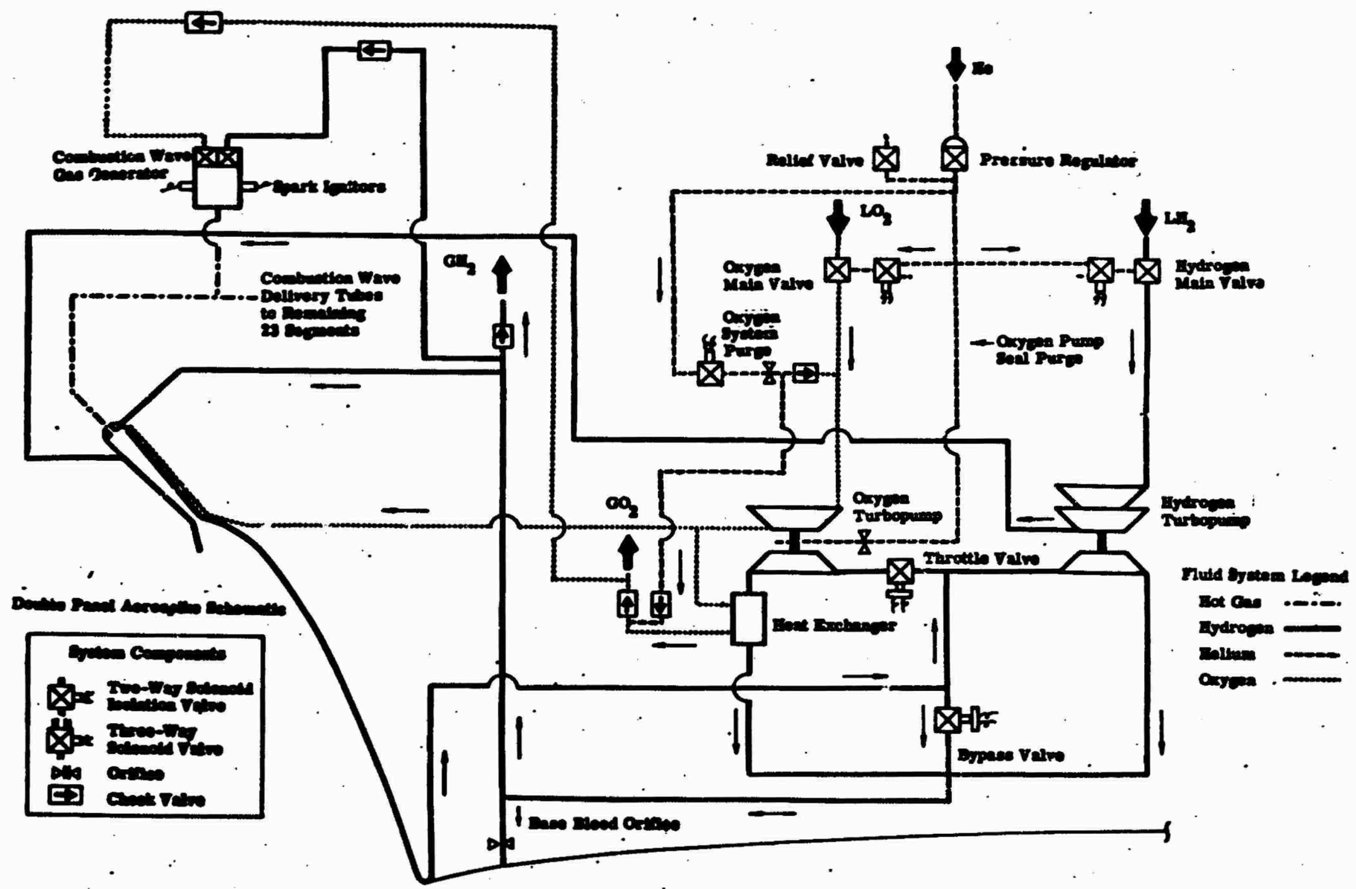


Figure B-1. Engine System Schematic

Preflight checkout was assumed to be the regular 2-hour or 60-start inspection and refurbishment for determining the failures that would be detected. Many of the failures, however, such as failure of a valve to actuate, would be identified by an automated on-board engine readiness checkout system which would monitor the engine condition before each mission.

During the time the AMPT engine is being transported in the Earth-to-Orbit Shuttle, it was assumed that OOS stage prevalves would be closed. The AMPT engine, therefore, has no function to perform so this phase of the mission was not considered.

The three phases of engine operation (start, mainstage, and cutoff) were considered separately. Mainstage includes not only steady-state operation at the design thrust and mixture ratio but, also, throttling and mixture ratio excursions.

Orbital coast accounts for those periods during a mission between engine burns. The stage is in orbit with fluids aboard.

RESULTS OF THE ANALYSIS

The engine system component breakdown used in this analysis is presented in Table B-1. The components were divided into four major categories: engine subsystems, control system, turbopump assemblies, and thrust chamber assembly. A table was prepared for each component category. In Tables B-2 through B-5, each individual component, as listed in Table B-1, is described and its failure modes and effects discussed. The designs of the various components did not provide sufficient detail to identify all possible failure modes that might be peculiar to the individual designs or fabrication techniques. Some of the possible failure modes can be determined only through detailed engine system design and development experience. Those failure modes that are identified were determined primarily from previous analysis of similar component designs. The possible causes were determined by a logical functional review of the design layout drawings.

TABLE B-1. AEROSPIKE ENGINE COMPONENTS

1. ENGINE SUBSYSTEM

   Gimbal Bearing and Thrust Mount
   Propellant Discharge Ducts and Manifolds
   Turbine Supply Ducts and Manifolds
   Turbine Discharge Ducts and Manifolds
   Heat Exchanger

2. CONTROL SYSTEM

   Main Oxygen Valve
   Main Hydrogen Valve
   Bypass Valve
   Throttle Valve
   Pneumatic Pressure Regulator
   Low-Pressure Relief Valve
   Pneumatic System Solenoid Isolation Valve
   Purge Solenoid Valves
   Purge Check Valves
   Pressurant Check Valves

3. TURBOPUMP ASSEMBLIES

   Oxygen Turbopump Assembly
   Hydrogen Turbopump Assembly

4. THRUST CHAMBER ASSEMBLY

   Injectors
   Combustion Chambers and Backup Structure Rings
   Nozzle and Base Closure
   Combustion Wave Igniter System

TABLE B-2. ENGINE SUBSYSTEM FMEA

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Gimbal Bearing & Thrust Mount	The thrust mount supports the engine system components. A spherical-type gimbal bearing transmits the engine thrust from the thrust mount to the vehicle & provides for thrust chamber gimbal displacement.	No applicable failure modes						
Propellant Discharge Ducts & Manifolds	These high pressure discharge ducts & manifolds carry the propellants from the exits of the pumps to the thrust chamber cooling passages.	No applicable failure modes						The ducting & manifold assemblies will be of all welded construction.
Turbine Supply Ducts & Manifolds	These high pressure ducts carry hot hydrogen gas from the exit of the thrust chamber cooling passages to the turbines & bypass valve. A throttle valve is located in the oxidizer turbine supply duct.	No applicable failure modes						The ducting & manifold assemblies will be of all welded construction.
Turbine Discharge Ducts & Manifolds	These high pressure ducts & manifolds carry hot hydrogen gas from the turbine exits & the bypass valve to the injectors &, thru an orifice, to the base closure for base bleed.	No applicable failure modes						The ducting & manifold assemblies will be of all welded construction.
Heat Exchanger	The heat exchanger consists of coils in the oxidizer turbopump exhaust duct. Its function is to convert liquid oxygen to gaseous oxygen for use as a stage tank pressurant.	No applicable failure modes						The heat exchanger installation will utilize all welded construction.

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. CONTROL SYSTEM FMEA

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT: ENGINE	PROBABLE FAILURE EFFECT: VEHICLE	PROBABLE FAILURE EFFECT: MISSION	REMARKS
Main Oxygen Valve	The main oxidizer valve is a normally closed, poppet-type valve operated by pneumatic system pressure. The valve is located between low-pressure propellant duct & engine oxidizer pump inlet. This component provides on/off control for the oxidizer flow.	Fails to open, opens partially, or opens erratically.	Electrical malfunction in solenoid pilot valve.	P	Detected & repaired during ground check.	None	Possible delay	
			Binding of pilot valve or main valve mechanism.	S	Insufficient oxidizer pressure detected & engine cutoff signaled	None	Abort	
			Excessive helium leakage around actuator piston.					
		Fails to close, closes partially, or closes erratically.	Electrical malfunction in solenoid pilot valve.	P	Detected & repaired during ground check.	None	Possible delay	
			Binding of pilot valve or main valve mechanism.	C	LOX-rich cutoff could result in chamber or injector damage.	Possible fire	Probable mission abort if additional engine burns required	Programmed stage pre-valve closure immediately following cutoff was assumed which would reduce the danger of fire damage & prevent the loss of tank oxygen supply
			Failure of poppet closing spring.		Extended cutoff impulse	None.	Greater than targeted impulse	
		Excessive oxidizer leakage through valve.	Poppet seat damage or contamination.	P	Detected & repaired during ground check.	None.	Possible delay.	
			Defective poppet stem bellows seal.	C	Possible fire hazard or hard start because of accumulated oxidizer.	Possible fire.	Possible abort	
		Erroneous position indication.	Malfunction in electrical system.	P	Detected & repaired during ground check.	None.	Possible delay	
			Faulty microswitch.	S, M, C, O	None.	None.	None	Other sensors will confirm satisfactory valve operation during mission, so mission abort is not necessary.
Main Hydrogen Valve	The main fuel valve is a normally closed, poppet-type valve operated by pneumatic system pressure. The valve is located between the low-pressure propellant inlet duct & the engine hydrogen pump inlet. This component provides on/off control for the hydrogen flow.	Fails to open, opens partially, or opens erratically.	Electrical malfunction in solenoid pilot valve.	P	Detected & repaired during ground check.	None.	Possible delay.	
			Binding of pilot valve or main valve mechanism.	S	Insufficient fuel pressure detected & engine cutoff signaled.	None.	Abort.	
			Excessive helium leakage around actuator piston.		Attempted oxidizer-rich start could result in chamber or injector damage.	Possible fire	Abort.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. (Continued)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Main Hydrogen Valve (cont'd)		Fails to close, closes partially, or closes erratically.	Electrical malfunction in solenoid pilot valve	P	Detected & repaired during ground check.	None	Possible delay	
			Binding of pilot valve or main valve mechanism	C	Extended cutoff impulse	None	Probable mission abort if additional engine burns required. Greater than targeted impulse	Programmed stage prevalve closure immediately following cutoff was assumed which would prevent the loss of tank fuel supply
			Failure of poppet closing spring.					
		Excessive hydrogen leakage through valve	Poppet seat damage or contamination	P	Detected & repaired during ground check.	None	Possible delay	
			Defective poppet stem bellows seal.	O	None.	None.	None	
		Erroneous position indication.	Malfunction in electrical system.	P	Detected & repaired during ground check.	None	Possible delay	
			Faulty microswitch	S, M, C, O	None.	None.	None.	Other sensors will confirm satisfactory valve operation during mission, so mission abort is not necessary
Bypass Valve	The bypass valve is a variable position, ball-type valve actuated by an electric motor. It controls the flowrate of hot hydrogen gas through a line bypassing the turbines. This valve is used for thrust control during mainstage engine operation, to speed up engine start by directing all the flow initially to the turbine and to reduce turbine flow at cutoff by opening to maximum bypass.	Fails to open during start or fails to open during cutoff.	Defective actuation motor.	P	Detected & repaired during ground check.	None	Possible delay	
			Loss of electric power to actuation motor.	S	Irregular start & thrust overshoot. Cutoff would be initiated to prevent engine failure from overthrust operation.	None	Abort.	
			Excessive mechanical resistance in actuator.					
			Faulty controller system.	C	Extended engine cutoff transient	None.	Possible abort. Greater than targeted impulse	
		Opens early during start or opens early at cutoff.	Failure to close after completion of previous cutoff.	P	Detected & repaired during ground check.	None.	Possible delay.	
				S	Slow engine start.	None.	Probably none.	
			Faulty controller system	C	Early engine thrust reduction.	None.	Lower than targeted impulse.	
		Irregular response & failure to maintain position commands.	Faulty potentiometer for feedback loop.	P	Detected & repaired during ground check.	None.	Possible delay	
			Excessive mechanical resistance.	S C	Irregular start or cutoff transient; may damage engine.	Probably none.	Uncertain, possible abort.	
				M	Irregular response to changes in thrust or mixture ratio.			
		Excessive internal leakage.	Faulty ball seal.	P	Detected & repaired during ground check.	None	Possible delay	
			Faulty ball seal bellows.	S	Slightly slow start.	None.	None.	
				M, C	None.	None	None.	
		Excessive external leakage.	Defective K-seal.	P	Detected & repaired during ground check.	None.	Possible delay.	
			Defective actuator shaft bellows seal.	S, M, C	Fire hazard.	Probably none	Probably none	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. (Continued)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Throttle Valve	The throttle valve is a variable position, ball-type valve actuated by an electric motor. It controls the flow of hot hydrogen gas to the oxidizer turbine. It is used in providing mixture ratio control during mainstage & in obtaining fuel-rich start & cutoff transients.	Fails to open during start or fails to close during cutoff.	Defective actuator motor	P	Detected & repaired during ground check.	None	Possible delay.	
			Loss of electrical power	S	Engine will not start.	None.	Abort.	
			Excessive mechanical resistance in actuator drive. Faulty controller system.	C	Engine cutoff transient will be extended; may damage T/P bearings	None.	Restart may not be possible. Greater than targeted impulse.	
		Opens early during start or closes early at cutoff.	Faulty controller system.	P	Detected & repaired during ground check.	None.	Possible delay.	
				S	May cause oxidizer T/P overspeed & damage. Possible overshoot in thrust & mixture ratio. Possible T/C damage.	None.	Possible abort.	
				C	Early cutoff.	None.	Lower than targeted impulse	
		Irregular response & failure to maintain position commands.	Faulty potentiometer for feedback loop.	P	Detected & repaired during ground check.	None.	Possible delay	
			Excessive mechanical resistance.	S, C	Irregular start or cutoff transient; may damage engine	Probably none.	Uncertain; possible abort.	
				M	Irregular response to changes in thrust or mixture ratio.			
		Excessive internal leakage.	Faulty ball seal.	P	Detected & repaired during ground check.	None.	Possible delay	
			Faulty ball seal bellows	S	Possible rotation of dry oxidizer pump; pump bearing damage.	None.	Possible abort	
				M	None.	None.	None.	
				C	Extended cutoff transient; may damage T/P bearings.	None.	Restart may not be possible. Greater than targeted impulse	
		Excessive external leakage.	Defective K-seal	P	Detected & repaired during ground check.	None	Possible delay.	
			Defective actuator shaft bellows seal.	S, M, C	Fire hazard.	Probably none	Probably none.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. (Continued)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Pneumatic Pressure Regulator	This regulator controls helium downstream pressure to 750 ±50 psig with an inlet pressure from 3600 to 900 psia.	High output	Ruptured bellows. Leaking poppet valve. Plugged compensating pressure port.	P	High output pressure detected & system repaired during ground check.	None.	Possible delay	
				S, M, C	High pressure would cause relief valve to open. Continued high pressure would cause helium depletion.	Deplete ambient helium supply.	Possible abort.	
		Low output.	Plugged inlet filter. Plugged compensating pressure port.	P	Low output pressure detected & system repaired during ground check.	None.	Possible delay	
				S	Failure of engine to start because of low valve actuation pressure.	None	Abort.	
				M	Premature engine shutdown.	None.	Abort.	
Low-Pressure Relief Valve	This relief valve is located directly downstream of the pressure regulator. The valve is designed to start to open at 825 psig & be fully open at 900 psig. This will ensure that the downstream pressure will not exceed 900 psig if the regulator fails.	Fails to open & relieve.	Ruptured bellows.	P	Detected & repaired during ground check.	None	Possible delay.	Relief valve actuation required only in the event of regulator failure so failure to open or failure to close would require a double failure.
				S, M, C	Could cause line rupture &/or seal damage.	Deplete helium supply.	Possible abort.	
		Fails to close after relieving.	Broken spring.	P	Detected & repaired during ground check.	None.	Possible delay	
				S, M, C	Could cause depletion of helium supply & inoperative pneumatic system.	Deplete helium supply.	Possible abort	
		Excessive leakage	Damaged poppet seat.	P	Detected & repaired during ground check.	None	Possible delay	
			Contamination	S, M, C	Could cause depletion of helium supply & inoperative pneumatic system.	Deplete helium supply.	Possible abort.	
Pneumatic System Isolation Valve	This valve provides a means of shutting off the helium flow to the engine pneumatic system.	Fails to open.	Faulty solenoid	P	Detected & repaired during ground check.	None	Possible delay	
				S	Engine would not start.	None.	Abort.	
		Opens prematurely	Faulty electrical controller.	P	Detected & repaired during ground check.	None.	Possible delay.	
				S	Oxidizer pump seal purge begins prematurely.	Slight excess use of helium.	Probably none.	
		Fails to close	Faulty solenoid.	P	Detected & repaired during ground check.	None.	Possible delay	
				C, O	Helium would escape through oxidizer pump seal purge.	Eventual loss of pneumatic helium supply.	Possible abort if additional starts required.	
		Excessive through leakage.	Damaged poppet.	P	Detected & repaired during ground check.	None.	Possible delay.	
			Contamination	C	Helium would escape through oxidizer pump seal purge.	Eventual loss of pneumatic helium supply.	Possible abort if additional starts required.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. (Continued)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Oxidizer System Purge Solenoid Valve	This 2-way solenoid valve controls the purge gas flow to the engine oxidizer system during start & cutoff.	Fails to open.	Faulty solenoid	P	Detected & repaired during ground check.	None	Possible delay	
				S	Fuel may enter oxidizer system & combust when oxidizer enters, causing engine damage.	Possible fire	Possible abort	
				C	Oxidizer may enter combustion chamber after fuel valve is closed, causing transient high mixture ratio condition with thrust chamber damage.	None	Possible abort if additional start required.	
		Opens prematurely	Faulty electrical controller.	P	Detected & repaired during ground check	None	Possible delay	
				S	Pneumatic system isolation valve prevents helium purge from beginning prematurely.	None.	None.	
				M, C	Purge injection pressure is higher than oxidizer tank pressure; therefore, the purge gas could stop the oxidizer flow & cause the helium purge to back up into the oxidizer tank. Helium entering the oxidizer pump could cause cavitation & pump overspeed.	Helium could be forced back into the main oxygen tank.	Possible abort.	
		Fails to close	Faulty solenoid	P	Detected & repaired during ground check.	None	Possible delay.	
				S	Possible oxidizer pump damage from helium dilution of oxidizer	Helium could be forced back into the main oxidizer tank.	Abort.	
				C	None; pneumatic system isolation valve would provide shutoff.	None.	Engine could not be successfully restarted unless valve closes. Mission abort if a restart is required.	
		Closes prematurely.	Faulty electrical controller	P	Detected & repaired during ground check.	None.	Possible delay.	
				S, C	Same as failure to open.	Same as failure to open.	Same as failure to open.	
		Excessive thru leakage.	Damaged poppet.	P	Detected & repaired during ground check.	None.	Possible delay	
			Contamination	M	Helium would leak into engine oxidizer flow.	Excess use of helium during engine operation; pneumatic system isolation valve would provide shutoff during orbital coast	Possible abort if helium supply exhausted.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. (Continued)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT: ENGINE	PROBABLE FAILURE EFFECT: VEHICLE	PROBABLE FAILURE EFFECT: MISSION	REMARKS
Fuel System Purge Solenoid Valve	This 2-way solenoid valve controls the purge gas flow to the engine fuel system during cutoff.	Fails to open.	Faulty solenoid	P	Detected & repaired during ground check	None	Possible delay	
				C	Residual hydrogen in the thrust chamber would not be expelled	None	Slight thrust from evaporating residual hydrogen.	Fuel purge may not be necessary on flight engine
		Opens prematurely.	Faulty electrical controller	P	Detected & repaired during ground check.	None	Possible delay	
				M, C	Purge injection pressure is higher than fuel tank pressure; therefore, the purge gas could stop the fuel flow & cause the helium purge to back up into the fuel tank. Helium entering the fuel pump could cause cavitation & pump overspeed. High T/C mixture ratio & T/C damage could occur.	Helium could be forced back into the main fuel tank.	Possible abort.	
		Fails to close	Faulty solenoid	P	Detected & repaired during ground check.	None.	Possible delay	
				C	None; pneumatic system isolation valve would provide shutoff.	None.	Engine could not be successfully restarted unless valve closes. Mission abort if a restart is required	
		Closes prematurely	Faulty electrical controller	P	Detected & repaired during ground check.	None	Possible delay	
				C	Insufficient purge; some residual hydrogen would remain.	None	Slight thrust from evaporating residual hydrogen.	
		Excessive through leakage.	Damaged poppet	P	Detected & repaired during ground check.	None.	Possible delay.	
			Contamination	M	Helium would leak into engine fuel flow.	Excess use of helium during engine operation; pneumatic system isolation valve would provide shutoff during orbital coast.	Possible abort if helium supply exhausted.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-3. (Concluded)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT: ENGINE	PROBABLE FAILURE EFFECT: VEHICLE	PROBABLE FAILURE EFFECT: MISSION	REMARKS
Purge Check Valves (2 required)	The check valves are located in the propellant purge lines downstream of the purge solenoid valves. The check valves protect the solenoid valves from exposure to the respective propellants.	Fails to open.	Excessive mechanical resistance.	P	Detected & repaired during ground check.	None	Possible delay	
				B, C	Engine purge gas will not flow; loss of oxidiser purge could cause engine damage.	Possible fire if loss of oxidiser purge	Possible abort	
		Fails to close or excessive reverse flow leakage	Excessive mechanical resistance. Damaged poppet. Contamination.	P	Detected & repaired during ground check.	None.	Possible delay.	
				M	Propellants are allowed to flow back up to system purge solenoid valves. No damage if purge solenoid valves are operating satisfactorily.	None.	None.	
Pressurant Check Valves (2 required)	The check valves are located in the propellant tank pressurization lines. These check valves prevent propellant from entering the engine during orbital coast.	Fails to open.	Excessive mechanical resistance.	P	Detected & repaired during ground check.	None.	Possible delay	
				M	Tank pressurization gas will not flow; insufficient pump inlet pressure will cause pump cavitation & engine shutdown.	No tank pressurization.	Abort	
		Fails to close or excessive reverse flow leakage	Excessive mechanical resistance Damaged poppet. Contamination.	C	Gas in tank pressurization lines will vent through engine after cutoff. Possible fire from venting gas.	None	Probably none	A pressurant shutoff valve in the vehicle is assumed to be closed during orbital coast.

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; C = orbital coast

TABLE B-4. TURBOPUMP ASSEMBLIES FMEA

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT: ENGINE	PROBABLE FAILURE EFFECT: VEHICLE	PROBABLE FAILURE EFFECT: MISSION	REMARKS
Oxygen turbopump assembly.	The oxidizer turbopump is a centrifugal pump with direct turbine drive & is equipped with a seal purge & drain system. Its function is to increase the pressure of the oxidizer & propel the fluid thru the high-pressure ducting to the chamber. The turbine is a single-row design & is driven by hydrogen gas from the thrust chamber regenerative cooling jacket. The hot hydrogen from the turbine exit is ducted to the thrust chamber injectors.	Failure to start	High torque because of frozen or misaligned bearing, shaft or seal	P	Detected & repaired during ground check.	None	Possible delay	
			Oxygen turbine throttle valve closed.	S	Engine will not start.	None	Abort	
		Fails to achieve design output in specified time.	High torque. Bypass valve open. Excessive downstream oxidizer resistance.	S, M	Engine may not start or may not achieve desired thrust &/or mixture ratio.	None.	Probable abort	
		Fails to remain in operation during mainstage or underspeed.	Bearing failure. Turbine erosion. Turbine gas supply leakage. Turbine throttle valve closes prematurely. Oxidizer explosion from rubbing of inducer or impeller, or contamination. Turbopump damage from axial thrust variation due to contamination in balance cavity orifices.	M	Engine will shutdown prematurely with possible damage to oxidizer pump Possible fire or explosion.	Probably none Possible fire damage if oxidizer explosion.	Probable abort	
		Overspeed	Loss of liquid oxidizer input. Ruptured high-pressure duct. Malfunction of engine start sequence or mainstage controller. Malfunction of overspeed cutoff network. Bypass valve fails to open during start or closes during mainstage.	S, M	Possible damage to oxidizer turbopump. Possible spike in chamber pressure with subsequent structural damage to thrust chamber. Possible thrust chamber damage from high mixture ratio operation	Probably none	Probable abort	Overspeed cutoff network should prevent serious damage.

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-4. (Continued)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Oxygen Turbopump Assembly (contd)		Internal shaft seal package leakage	Contamination.	P	Detected & repaired during ground check.	None.	Possible delay	
			Defective or damaged seals	M	Loss of performance	None.	Possible failure to achieve mission goals due to propellant depletion from low performance or engine shutdown due to malfunction detection.	
					Possible fire or explosion in seal cavity due to hydrogen & oxygen mixing.	Possible fire damage.	Abort.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-4. (Concluded)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT: ENGINE	PROBABLE FAILURE EFFECT: VEHICLE	PROBABLE FAILURE EFFECT: MISSION	REMARKS
Hydrogen Turbopump Assembly	The fuel turbopump is a centrifugal pump with direct turbine drive. Its function is to increase the pressure of the fuel and propel the fluid through the cooling jacket to the injector. The turbine is a single row design & is driven by hydrogen gas from the thrust chamber regenerative cooling jacket. The hot hydrogen from the turbine exit is ducted to the thrust chamber injectors.	Failure to start.	High torque because of frozen or misaligned bearing, shaft, or seal	P S	Detected & repaired during ground check. Engine will not start.	None. None.	Possible delay. Abort.	
		Fails to achieve design output in specified time.	High torque. Bypass valve open. Excessive downstream fuel resistance. Excessive turbine flow resistance.	S, M	Engine may not start or may not achieve desired thrust &/or mixture ratio. Thrust chamber damage may occur from higher than nominal mixture ratio.	None.	Probable abort.	
		Fails to remain in operation during mainstage or underspeed.	Bearing failure. Turbine erosion. Turbine gas supply leakage. Turbopump damage from axial thrust variation due to contamination in balance cavity orifice.	M	Engine will shut down prematurely with certain damage to the thrust chamber due to lack of coolant flow. Possible damage to hydrogen turbopump.	None.	Probable abort.	
		Overspeed.	Loss of liquid fuel input. Ruptured high-pressure duct. Malfunction of engine start sequence or mainstage controller. Malfunction of overspeed cutoff network. Bypass valve fails to open during start or closes during mainstage. Oxidizer turbine throttle valve closes during mainstage.	S, M	Possible damage to fuel turbopump. Possible spike in chamber pressure with subsequent structural damage to thrust chamber.	Probably none.	Probable abort.	Overspeed cutoff network should prevent serious damage.
		Internal shaft seal package leakage.	Contamination Defective or damaged seals	M	Loss of performance.	None.	Possible failure to achieve mission goals due to propellant depletion from low performance or engine shutdown due to malfunction detection.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-5. THRUST CHAMBER ASSEMBLY FMEA

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Injectors	The injectors accept gaseous hydrogen from the turbine exhaust & the gaseous oxygen from the oxidizer coolant jacket & inject them into the combustion chamber in a manner resulting in complete stable combustion.	Injector orifice obstruction	Contamination	S, M, C	Probably none. If obstruction is severe enough, performance degradation & damage from local mixture ratio shift are possible.	Probably none.	Probably none. Possible propellant depletion or abort due to damaged chamber assembly.	
Combustion Chambers	The combustion chambers provide a combustion zone for the propellants & provide for an initial expansion of the propellants. The chambers are regeneratively-cooled cast segments with a backup oxidizer coolant circuit on the inner wall. Two continuous structural rings provide inner & outer support for the segments.	Slight internal leakage	Thermal fatigue or material deficiency	P S, M, C	Detected & repaired during ground check Probably none. If severe enough performance degradation, mixture ratio shift & thrust misalignment are possible.	None Probably none	Possible delay Probably none Possible propellant depletion.	
Nozzle & Base Closure	The regeneratively-cooled tubular nozzle provides an expansion surface for the combustion gases from the combustors. The base closure distributes the base bleed flow into the base cavity. Thrust from the base & nozzle is transmitted into the combustion chamber assembly.	Slight external leakage in nozzle	Thermal fatigue or material deficiency	P S, M, C	Detected & repaired during ground check. Probably none If severe enough performance degradation, mixture ratio shift & thrust misalignment are possible.	None. Probably none.	Possible delay Probably none Possible propellant depletion.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

TABLE B-5. (Concluded)

COMPONENT	FUNCTION	POSSIBLE TYPE OF FAILURE	POSSIBLE FAILURE CAUSE	SEQ.	PROBABLE FAILURE EFFECT			REMARKS
					ENGINE	VEHICLE	MISSION	
Combustion Wave Igniter System	This ignition system provides the pilot flames for igniting the combustion chamber segments. The system consists of a premix chamber, dual spark igniters for the premix chamber, 6 igniter elements for each combustor. Propellants in the premix chamber are ignited causing a combustion wave to travel to each of the igniter elements & igniting the propellant flowing thru the element. This element provides the pilot flame for the combustor.	Premix chamber O_2 solenoid valve fails to open.	Faulty solenoid.	P	Detected & repaired during ground check.	None.	Possible delay.	
			Faulty electrical controller.	S	Engine will not ignite. Insufficient system pressures detected & engine cutoff signaled.	None.	Abort.	
		Premix chamber O_2 solenoid valve fails to close.	Faulty solenoid.	P	Detected & repaired during ground check.	None.	Possible delay.	
			Faulty electrical controller	S	Possible fire or explosion in premix chamber or premix lines from continued oxidizer flow. Igniter element damage from high mixture ratio operation.	Probably none.	Abort due to engine shutdown from performance degradation.	
		Igniter element failure (same failures as other injector elements)						
		Premix spark igniter fails to provide spark.	Stage power supply failure.	P	Detected & repaired during ground check.	None.	Possible delay.	Would require double failure since redundant spark igniters are used.
			Internal electrical failure in plug/exciter unit.	S	Engine will not start.	None.	Abort.	
		Element flow fails to ignite.	Improper sequencing of spark. Flow blockage in element or premix chamber feed lines from contamination &/or handling damage.	S	Engine probably will not complete start. Insufficient system pressures will be detected & engine cutoff signaled. If only part of the segments ignite, there is a possible fire or explosion danger from unlit segment propellants.	Possible fire damage.	Abort.	

P = preflight checkout; S = start sequence; M = mainstage; C = cutoff; O = orbital coast

APPENDIX C

DOUBLE-PANEL ENGINE TURBOPUMP
DESIGN CONFIGURATION SELECTION

DESIGN CRITERIA

The critical performance requirements of the turbopumps as dictated by engine considerations are presented in Table C-1. The preliminary analysis and tradeoff studies were accomplished on the basis of the initial engine balance included in the first two columns of Table C-1. Subsequently, design changes to the thrust chamber cooling circuit resulted in an increase in the discharge pressure requirements of the oxidizer and fuel pumps and a corresponding change in the inlet conditions for both turbines. The revised conditions are included in the third and fourth columns of Table C-1. The changes were not sufficient to invalidate the conclusions of the tradeoff studies. The final layouts of the turbopumps were based on the revised numbers.

The general design philosophy was to use existing technology. In addition to the performance requirements presented in Table C-1, the following criteria were applicable:

Time Between Inspections	:	2 hours, 60 starts
Time Between Overhaul	:	10 hours, 300 starts
Total Life	:	50 hours, 1500 starts
Inspection	:	≤5 percent of original cost
Overhaul	:	≤25 percent of original cost

CONFIGURATION SELECTION

To meet the performance requirements of the engine as listed in Table C-1, a survey of potential configurations was made. Specific speed $(= \frac{NQ^{1/2}}{H^{3/4}})$ values and the

TABLE C-1. AMPT AEROSPIKE LOX/LH_2 ENGINE TURBOPUMP PERFORMANCE REQUIREMENTS

o **ENGINE INFORMATION**				
Type	Aerospike			
Thrust	25000 lb			
Chamber Pressure	1000 psia			
Nozzle Area Ratio	200			
Engine Mixture Ratio	5.5			
Secondary Mixture Ratio	0.0			
Turbine Drive Cycle	Expander Topping Cycle			
Turbine Arrangement	Parallel			
	Initial Engine Balance Used for Trade Off		Engine Balance Used for Final Layout	
o **PUMP REQUIREMENTS**	FUEL	OXIDIZER	FUEL	OXIDIZER
Fluid	LH_2	LOX	LH_2	LOX
Flowrate, lb/sec	8.164	44.903	8.164	44.903
Inlet Pressure, psia	15	25	15	25
Discharge Pressure, psia	3155	1466	3270	1736
NPSH, ft	60.0	16.0	60.0	16.0
o **TURBINE REQUIREMENTS**				
Fluid	GH_2	GH_2	GH_2	GH_2
Inlet Temperature, ^oR	1146	1146	1001	1001
Inlet Pressure, psia	1943	1838	1977	1712
Exhaust Pressure, psia	1190	1195	1190	1195
o **THROTTLING REQUIREMENT**	5:1			

goal of minimum weight eliminated positive displacement pumps from consideration for both fluids. Although the specific speed per stage would have approached reasonable values for an axial-flow fuel pump, this concept was discarded because it has undesirable stall characteristics which complicate engine start and would require very small blade sizes. As a result, the design effort was restricted to centrifugal types for both pumps. Axial flow impulse turbines were selected (as opposed to radial turbines) because at the low velocity ratios they offer better efficiency, and they have higher stall torque and lower inertia.

Performance figures were generated for a two- and a three-stage fuel pump, each powered by a single- or two-row turbine. For the oxidizer turbopump, a single-stage pump configuration was analyzed with single-row turbine. The use of the fuel turbine first stage modified for partial admission also was evaluated for the oxidizer turbopump. A higher speed version of each turbopump also was investigated to illustrate the advantages in performance and weight which could be realized in the event the net positive suction head (NPSH) levels could be raised above the values presently stipulated for the engine.

The parameters and performance values obtained for the fuel turbopump are presented in Table C-2. The operating speed was established by suction performance requirements at 75,000 rpm for both the two- and the three-stage pump configurations. The three-stage pump offered a higher efficiency by four percentage points (62 percent versus 58 percent) because of the higher specific speed per stage. Although the three-stage pump is a more complex design because of the additional parts involved, it is estimated to be cheaper because conventional, integrally cast impellers can be used. The tip speed required for the two-stage pump exceeds the limit for cast steel and aluminum so diffusion-bonded titanium construction would be required. Increasing the turbine pitch diameter from 4 to 5 inches yielded a substantially improved turbine efficiency because of the more favorable tip speed-to-gas spouting velocity ratio. Turbine blade stresses even at the increased diameter are well below the capability of high-strength turbine materials at the specified operating gas temperatures.

TABLE C-2. AMPT (MK-46) LH_2 TURBOPUMP

PERFORMANCE OF CANDIDATE CONFIGURATIONS
(Based on flow and pressure requirements
of initial engine systems analysis)

PUMP:		
No. of Pump Stages	2	3
$\dot{w}$ (lb/sec)	8.164	8.164
Q (gpm)	814	814
H (ft)	94500	94500
P_d (psia)	3155	3155
NPSH (ft)	60	60
S_s (rpm gpm $^{1/2}$/ft$^{3/4}$)	99000	99000
N (rpm)	75000	75000
D_{imp} (in)	5.6	4.3
u_{imp} (fps)	1835	1408
(N_s) Stg	605	840
η (%)	58	62
BHP	2420	2260
Impeller Type	Diffusion Bonded Ti	Cast Steel or Al

TURBINE:						
No. of Rows	1	2	1	1	2	1
Admission (%)	100	100	100	100	100	100
η (%)	56	64	67	56	64	67
$\dot{w}$ (lb/sec)	5.82	5.03	4.9	5.43	4.7	4.58
D_m (in)	4.0	4.0	5.0	4.0	4.0	5.0
u (fps)	1310	1310	1640	1310	1310	1640
$AAN^2 \times 10^{-9}$ ($in^2 rpm^2$)	27.3	28.3	26.5	25.5	26.9	24.5
u/C_o	.255	.310	.32	.255	.310	.32
Blade height	.387	.400	.300	.360	.380	.280

In Table C-3, the performance parameters for the oxidizer turbopump are presented. Here again, the operating speed was established by the NPSH criteria at 22,000 rpm. The possibility of using a modified fuel turbine for the oxidizer turbine was explored. Although this approach would result in a notable savings in cost and weight, the required turbine weight flow would greatly increase (1.6 to 2.75 lb/sec).

In Table C-4, a definition of the turbopumps is provided with the available NPSH raised to 36 feet for the LOX pump and 88 feet for the LH_2 pump. The effect of the higher NPSH is to allow operating the LOX pump at 40,000 rpm and the LH_2 pump at 1000,000 rpm, resulting in a substantial weight savings (total weight was estimated at 30 pounds below lower speed versions with the two-stage LH_2 pump), and more compact design. The higher speed version appears particularly attractive for the LH_2 pump, where the increase from 60 to 88 feet of NPSH represents less than 1 psi.

The combinations of various fuel and oxidizer turbopump configurations is examined in Table C-5. Of primary significance in this table is the percent turbine bypass flowrate with each configuration. The bypass is the portion of LH_2 delivered by the pump which is not passed through the turbines, representing, in effect, the power or calibration margin of the engine. To provide a conservative margin. the target nominal bypass flowrate was set at 17 percent, with a lower limit of 10 percent. This criterion automatically eliminated several of the configurations in Table C-5 because the combined performance of the turbopumps resulted in an inadequate (in some instances) negative bypass. This process of elimination left as potential fuel turbopump candidates a two-stage pump with a two-row, 4-inch-diameter turbine or one-row, 5-inch-diameter turbine and a three-stage pump with either a one- or two-row turbine. The use of the fuel turbine first row for the oxidizer turbine was ruled out due to inadequate performance, leaving a single-stage pump configuration with a single-row turbine optimized for the oxidizer pump speed. The high-speed designs noted in the last column of Table C-5 require more NPSH than provided by the present ground rules and, as a result, represent only background information.

TABLE C-3. AMPT (MK-46) LOX TURBOPUMP PERFORMANCE OF CANDIDATE CONFIGURATIONS (Based on flow and pressure requirements of initial engine systems analysis)

PUMP:			
No. of Stages		1	
$\dot{w}$	(lb/sec)	44.903	
Q	(gpm)	283	
H	(ft)	2960	
P_d	(psia)	1466	
NPSH	(ft)	16	
S_s	(rpm gpm $^{1/2}$/ft $^{3/4}$)	46,000	
N	(rpm)	22,000	
D_{imp}	(in)	4.55	
u_{imp}	(fps)	437	
N_s	(rpm gpm $^{1/2}$/ft $^{3/4}$)	930	
η	(%)	68	
BHP		354	
TURBINE:			
No. of Rows		1	1 *
Admission	(%)	8.66	50
η	(%)	32	20.1
$\dot{w}$	(lb/sec)	1.6	2.75
D_m	(in)	8	4.0
u	(fps)	768	384
$AAN^2 \times 10^{-9}$	($in^2 rpm^2$)	7.2	2.0
u/C_o		.162	.087
Blade height		.590	.326

* Uses LH_2 turbine modified for partial admission

TABLE C-4. AMPT (MK-46) LH_2 AND LOX TURBOPUMPS PERFORMANCE WITH INCREASED AVAILABLE NPSH

		LH_2 T/P	LOX T/P
PUMP:			
No. of Pump Stages		2	1
$\dot{w}$	(lb/sec)	8.164	44.903
Q	(gpm)	814	283
H	(ft)	94500	2960
P_d	(psia)	3155	1466
NPSH	(ft)	88	36
S_s	(rpm gpm $^{1/2}$/ft $^{3/4}$)	100,000	46,000
N	(rpm)	100,000	40,000
D_{imp}	(in)	4.3	2.5
u_{imp}	(fps)	1878	437
(N_s) Stg.		805	1685
η	(%)	61	74
BHP		2300	328
Impeller Type		D.B. Ti	Cast Steel
TURBINE:			
No. of Rows		1	1
Admission	(%)	100	40
η	(%)	61.6	32
$\dot{w}$	(lb/sec)	5.05	1.11
D_m	(in)	3.5	3.5
u	(fps)	1525	612
$AAN^2 \times 10^{-9}$	(in^2 rpm^2)	44.5	3.5
u/C_o		.298	.128
Blade height (inch)		.405	.200

TABLE C-5. AMPT (MARK-46) TURBOPUM
(based on initia

LH_2 Pump Stages	2					
LH_2 Turbine Rows	1			2		
LH_2 Speed, rpm	75,000			75,000		
LH_2 NPSH, feet	60			60		
LH_2 Turbine Pitch Diameter, inches	4	4	5	4		
LOX Turbine Configuration	Optimized	LH_2 Turbine	Optimized	Optimized	First-Row LH_2 Turbine	Opti
LOX Speed, rpm	22,000	22,000	22,000	22,000	22,000	22,0
LOX NPSH, feet	16	16	16	16	16	16
Engine Turbine Bypass, flowrate	0.74	-0.406	1.7	1.53	0.384	1.13
Engine Turbine Bypass, percent	9	-5	21	19	4.7	14
Weight (estimate), pounds	Baseline	-9	+5	+5	-4	+13
Cost (estimate), dollars	Baseline	-\$5000	0	+\$10,000	+\$5000	-\$90
LH_2 Rotordynamics (estimate)						
Overhung Impellers	First Critical ≈ 30 K Second Critical ≈ 40 K			First Critical ≈ 25 K Second Critical ≈ 40 K		
Inboard Impellers	First Critical ≈ 20 K Second Critical ≈ 90 K Third Critical ≈ 100 K			First Critical ≈ 15 K Second Critical ≈ 90 K Third Critical ≈ 100 K		Firs Secc Thi

*D··iates from Air Force Requirement

)

5) TURBOPUMP CONFIGURATION SUMMARY
d on initial engine balance)

	3					2
	1			2		1
	75,000			75,000		100,000
	60			60		87.5*
	4	5	4	4		3.5
ow LH_2 ne	Optimized	Optimized	LH_2 Turbine	Optimized	First-Row LH_2 Turbine	LH_2 Turbine
	22,000	22,000	22,000	22,000	22,000	40,000
	16	16	16	16	16	35*
	1.13	2.00	-0.016	1.86	0.714	2.0
	14	25	0	23	9	25
	+13	+18	+4	+18	+9	-30
	-$9000	-$9000	-$14,000	+$1000	+$1000	-$5000
25 K						First Critical ≈ 40 K
≈ 40 K						Second Critical ≈ 45 K
15 K	First Critical ≈ 20 K			First Critical ≈ 15 K		First Critical ≈ 25 K
≈ 90 K	Second Critical ≈ 70 K			Second Critical ≈ 70 K		Second Critical ≈ 90 K
100 K	Third Critical ≈ 90 K			Third Critical ≈ 90 K		Third Critical ≈ 100 K

2

From the above acceptable fuel turbopump candidates, two configurations were chosen for more detailed study: the two-stage pump with a single-row, 5-inch-diameter turbine and the three-stage pump, also with a single-row, 5-inch-diameter turbine. Soft line sketches were made of these two configurations (Fig.C-1 and C-2) to obtain a concrete weight comparison. Results of the weight calculations showed no significant difference between the two designs. The explanation for this apparent paradox is that the diameter (which has a squared effect on weight) of the two-stage pump was larger and the pump length could not be reduced by the ratio of the stages because the higher tip speeds attendant with the two-stage pump require thicker impeller backplates.

As noted above, the efficiency of the three-stage fuel pump is higher and its manufacturing cost is lower. A further consideration was the fact that, although the feasibility of diffusion bonding has been demonstrated, production experience is not very extensive and, as a result, additional development effort may be required to perfect fabrication techniques.

In view of the above factors, the three-stage fuel pump appeared the superior configuration and was therefore, selected for hard line layout.

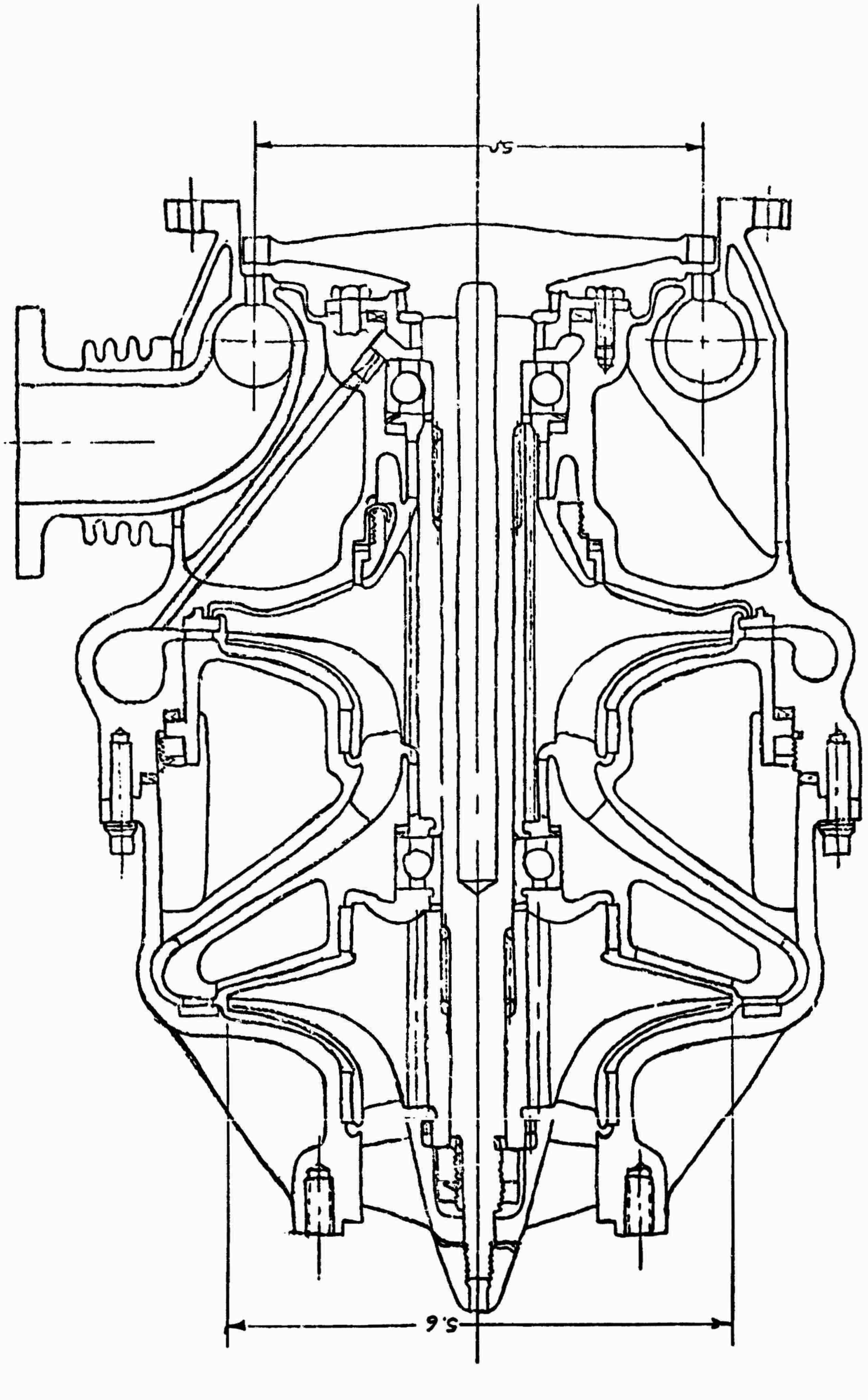


Figure C-1. AMPT Fuel Turbopump Two-Stage Pump
Preliminary Drawing Used for Configuration Selection

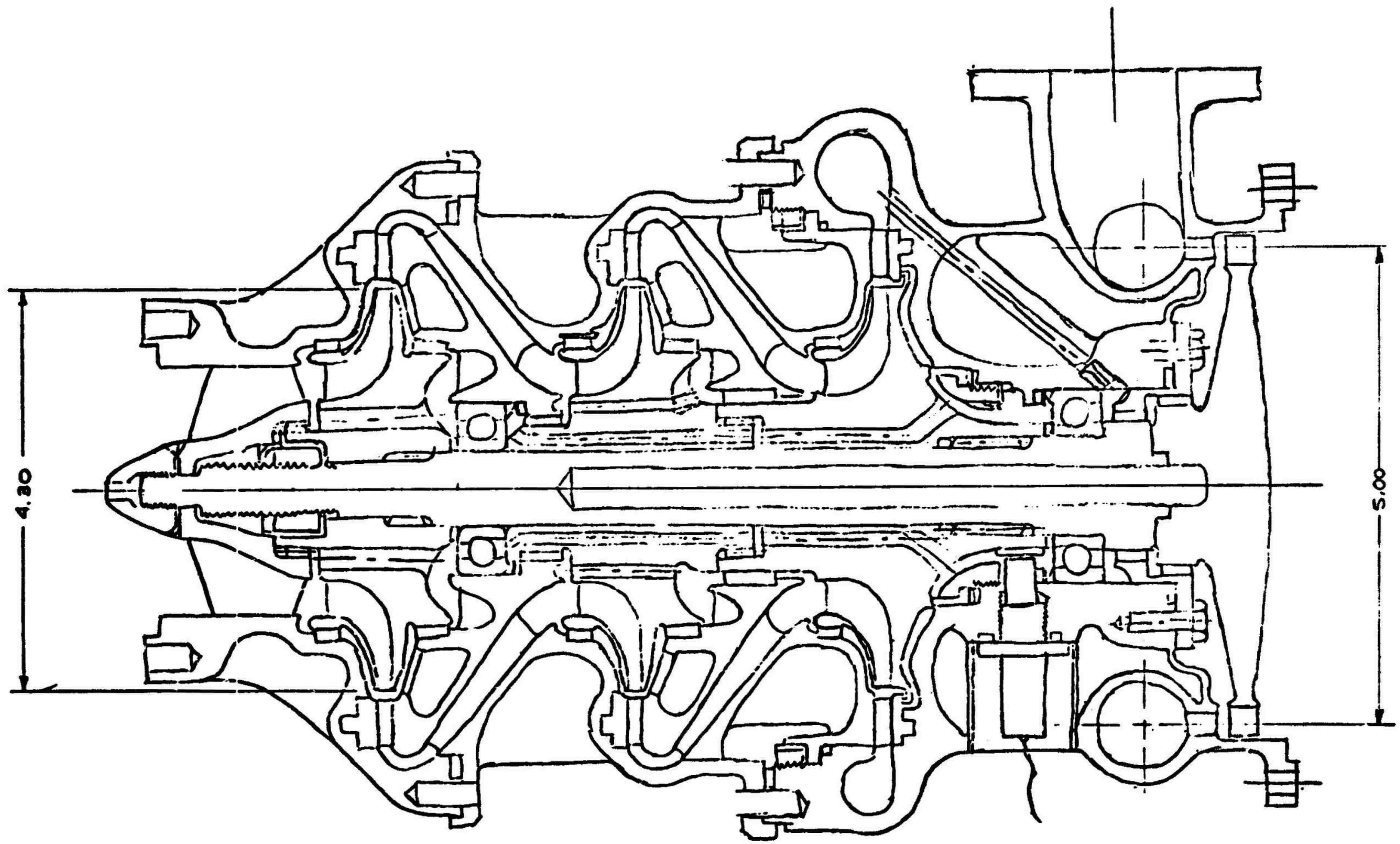


Figure C-2. AMPT Fuel Turbopump Three-Stage Pump
Preliminary Drawing Used for Configuration Selection

APPENDIX D

IGNITION SYSTEM SELECTION

Studies were conducted to select an ignition system for the AMPT aerospike engine. Suitable ignition methods were identified and assessed in terms of relative weight, design complexity, reliability, and technology status. The objective of achieving a reliable, lightweight ignition system as the culmination of a low-risk development effort was utilized as the basic selection guideline. Therefore, emphasis was directed to methods currently in use or having sufficient experimental verification to ensure success in an operational system.

The three most promising ignition techniques applicable to the AMPT aerospike thrust chamber are augmented spark, resonance, and combustion wave. These ignition systems are discussed in the following sections. Use of a hypergolic third propellant, such as fluorine or chlorine trifluoride, was eliminated because of toxicity and the maintenance and handling problems associated with reusability of the engine. Catalyst bed ignition was eliminated because of the multiple-start and long life requirements.

AUGMENTED SPARK IGNITER

The reliability of augmented spark igniters (ASI) has been proved in the J-2 engine. The configuration used in that engine, however, would be excessively large and heavy for the AMPT application where 24 separate ignition sources are required. A compact, integrated spark plug/exciter unit being developed for the Space Shuttle Auxiliary Propulsion System is better suited to the AMPT engine. However, even an ignition system using the integrated exciter and spark plug will result in a heavy system weight because each unit weighs nearly 1 pound. A central exciter with individual plugs could be utilized, but much of the weight savings is negated by the 50 feet of shielded, evacuated cable required. The cable also could aggravate any radio interference problems.

A flow schematic for an ASI with integrated spark plug/exciter units is presented in Fig.D-1 and a preliminary drawing of the method by which the ASI can be integrated into the chamber segment is shown in Fig.D-2. The ignition system includes an integrated spark plug/exciter (not shown in Fig. D-2) to provide an electrical arc discharge; an oxygen hydrogen feed system and injector to create a combustible mixture; a combustion chamber; and a concentric element to carry the ASI flow to the thrust chamber segment.

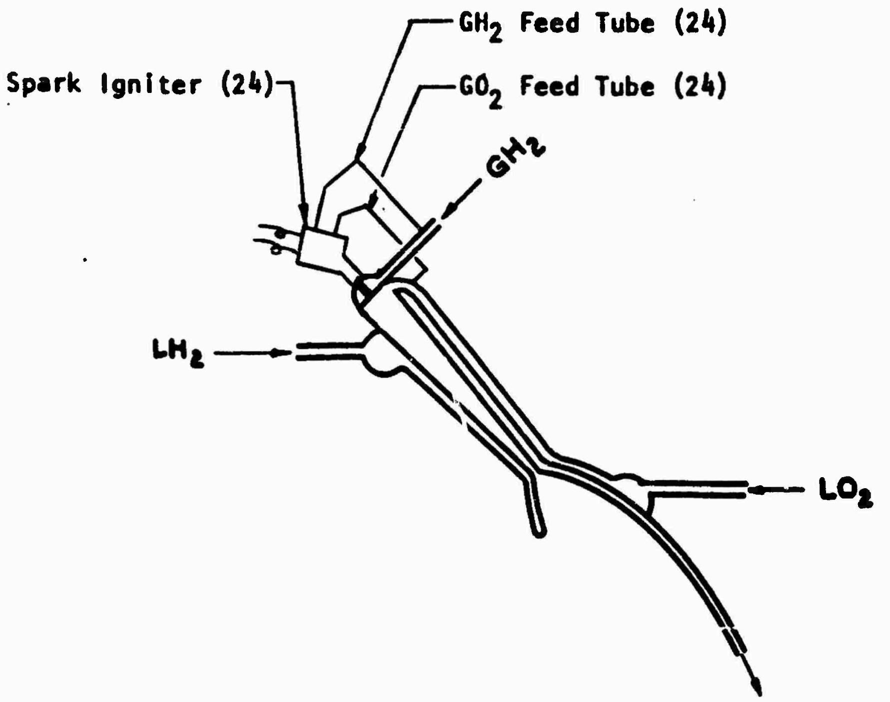
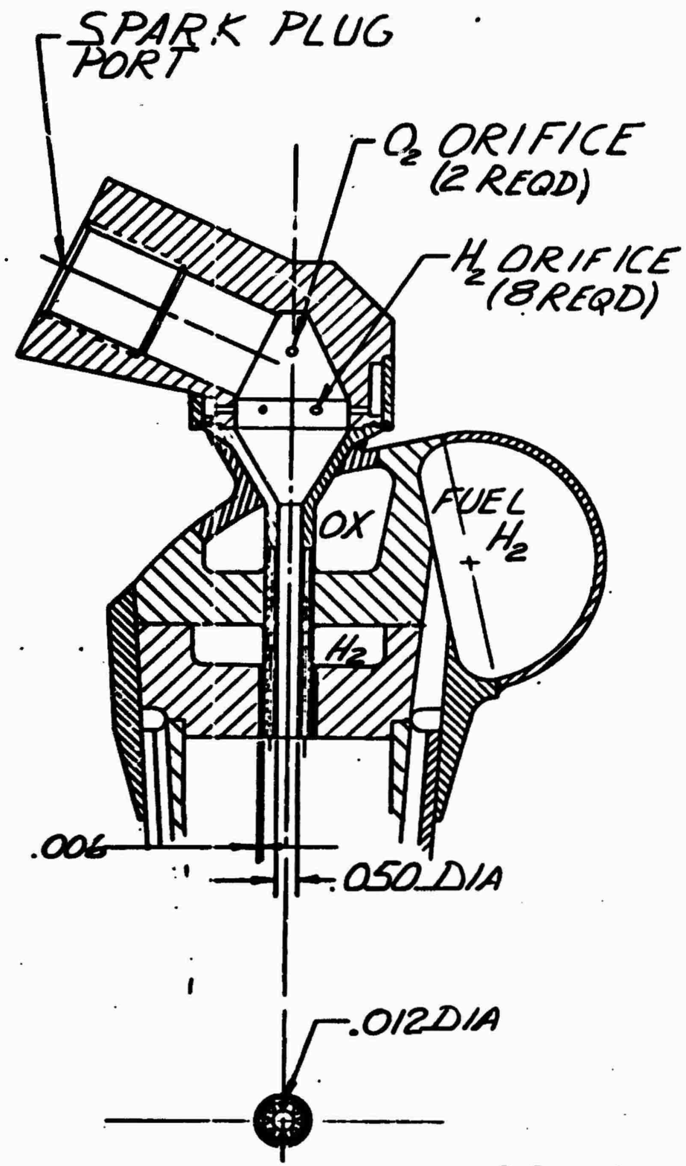
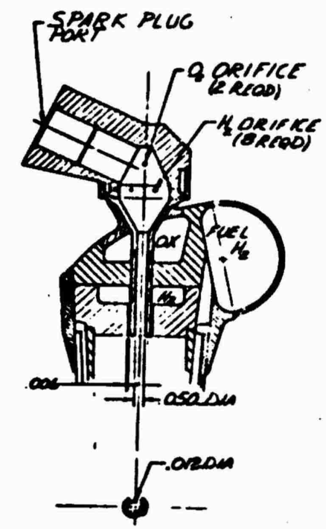


Figure D-1 Augmented Spark Igniter



2.5 x size



Full Size

Figure D-2. ASI Installation

Thrust chamber segment conditions at ignition were estimated from the engine start transient information as given in Table D-1.

TABLE D-1. ENGINE START TRANSIENT

Chamber Pressure, psia	10
Hydrogen Flowrate, lb/sec	0.00625
Oxygen Flowrate, lb/sec	0.014
Hydrogen Pressure, psia	39 } at the injector Manifolds
Oxygen Pressure, psia	34 } at the injector Manifolds

Mainstage engine P_c is 750 to 1000 psia.

Features of the design shown in Fig. D-2 are shown in Table D-2.

TABLE D-2. DESIGN FEATURES

Chamber Pressure, psia	25
Total Flowrate, lb/sec	0.0002
Mixture Ratio	1.5
Throat Diameter, in.	0.050
Oxygen Pressure, psia	34
Hydrogen Pressure, psia	39

The ASI design consists of a truncated cylindrical/conical body with eight tangential entry H_2 orifices in one plane and a 90-degree O_2 doublet impinging slightly upstream of the plane of the H_2 entry. The spark plug is located in the conical surface normal to the plane of the O_2 orifices and above the H_2 orifices. The design concentrates oxidizer near the centerline and surrounds it with swirling fuel to cool the ignition unit. The spark plug tip is located between the oxidizer and fuel injection planes and the integrated plug/ exciter extends toward the engine centerline. The combusted hot gas flows down the center of the igniter element and ignites the concentric

oxidizer and fuel pilot flows. These pilot flows are fed by the main injector manifolds and during mainstage operation provide an overall element mixture ratio equal to the thrust chamber design value. After ignition has been established, the plug/exciter unit is turned off.

The advantage of the ASI design is that it takes its propellants directly from the injector manifolds as shown by the design supply pressures. The ASI flows 1 percent of the injector flow at ignition. Concentric augmentation flow from the manifolds increases the element flow by 4 percent of the injector flow to ensure ignition. One percent flow has been shown to be adequate in the past but, due to the transient nature of the flow conditions, the triaxial flow element (hot-gas center, O_2 middle, H_2 outside) should provide a more positive ignition torch.

The primary disadvantage of the ASI design is the size of the spark plug exciter units required for each chamber. A minimum plug port size of 0.225 OD x 0.50 is shown, but the exciter would be at least 1.5 diameter x 2.0, which will dwarf the rest of the assembly. If a small electrical source were developed, the hot-gas device is small enough to be practical.

A variation of the ASI design which has just been described is shown in Fig. D-3 This concept is referred to as a plasma-type ASI. Oxidizer flows from an annular manifold, around the spark electrode where it is ionized, and into the igniter combustion chamber. This type of design provides cooling for the electrode and minimizes the erosion potential from combustion. A small portion of the hydrogen is injected from an annular manifold, impinges with the oxygen in the igniter chamber, and combusts hypergolically with the ionized oxygen. The remaining hydrogen flows down the annulus of the concentric element, cools the inner hot-gas tube, and, during mainstage operation, provides an element mixture ratio equal to the thrust chamber design value.

The weight of ASI units is competitive with a resonance ignition system or a combustion wave ignition system with gas-propellant accumulators, but is

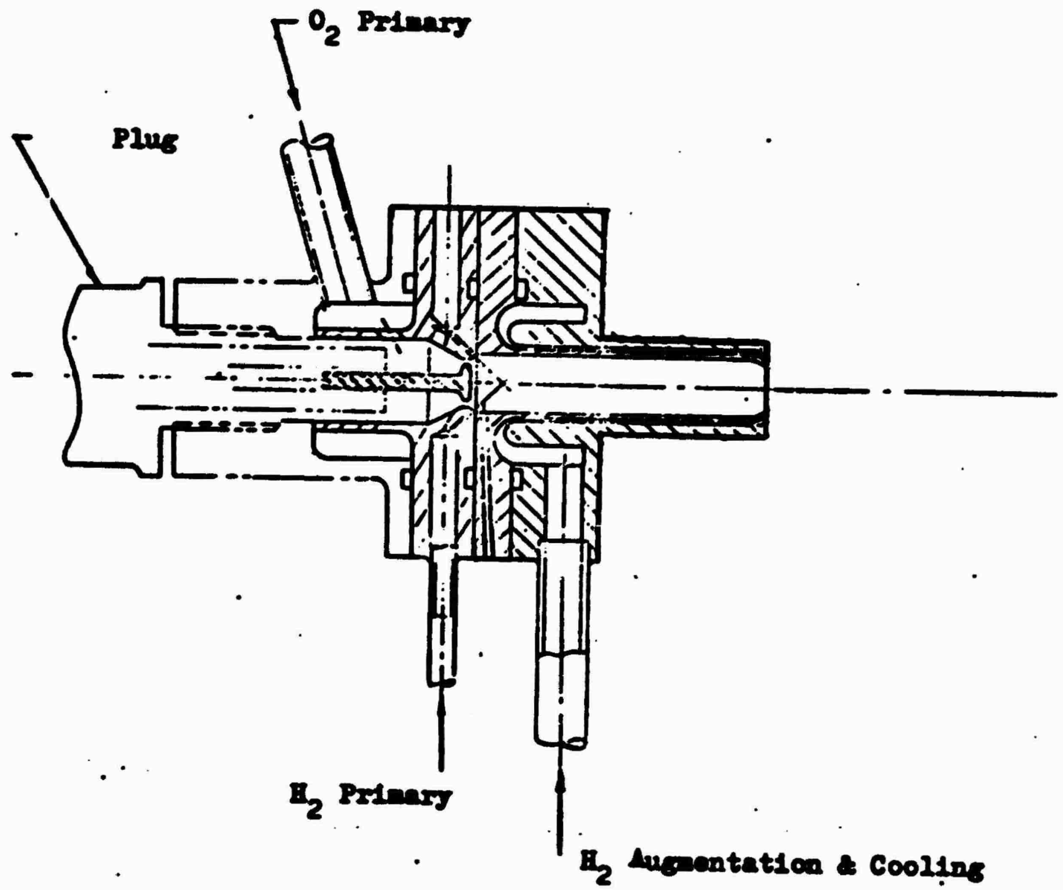


Figure D-3. Plasma-Type ASI

heavier than a combustion wave system without accumulators. The large size (approximately 2 inches in diameter) also is a distinct disadvantage. Because 24 units are required, they cannot be easily integrated into the thrust chamber/ injector assembly.

RESONANCE IGNITER

Although resonance ignition is not currently in use in operational rocket engines, the feasibility and reliability of this technique have been demonstrated in technology programs conducted in support of the Space Shuttle Auxiliary Propulsion System development. Most of the experience to date has been with ambient temperature propellants with a hydrogen pressure down to approximately 150 psia. Operation at other conditions appears feasible, although limits have not been investigated sufficiently. Thus, the present resonance igniter concept would require gas storage bottles for its supply of igniter propellants. The weight of these bottles is the primary disadvantage of this system.

A flow schematic for a typical resonance igniter is shown in Fig. D-4, and a preliminary design is presented in Fig. D-5. It has opposed tubes which serve as propellant inlets. Features of the design are given in Table D-3.

TABLE D-3. IGNITER DESIGN FEATURES

Chamber Pressure, psia	100
Total Flowrate, lb/sec	0.001
Mixture Ratio	1.0
Throat Diameter, in.	0.050
Oxygen Pressure, psia	1200 to 200
Hydrogen Pressure, psia	1200 to 200

The resonance igniter consists of a sonic H_2 driver nozzle, an inline resonance cavity with a pressure-actuated valve at the end, and a combustion chamber body. Hydrogen flows through the sonic nozzle into the resonance cavity where

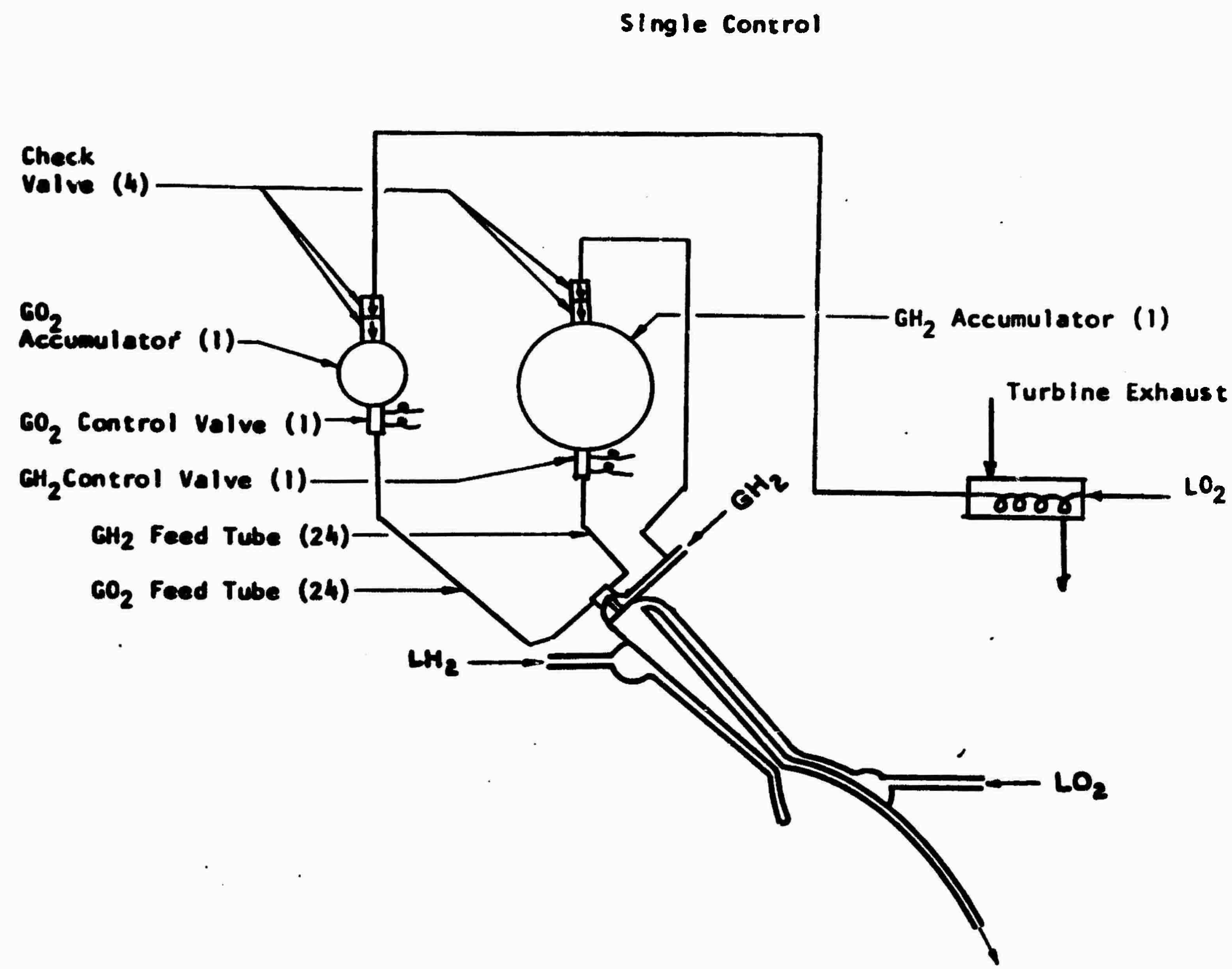
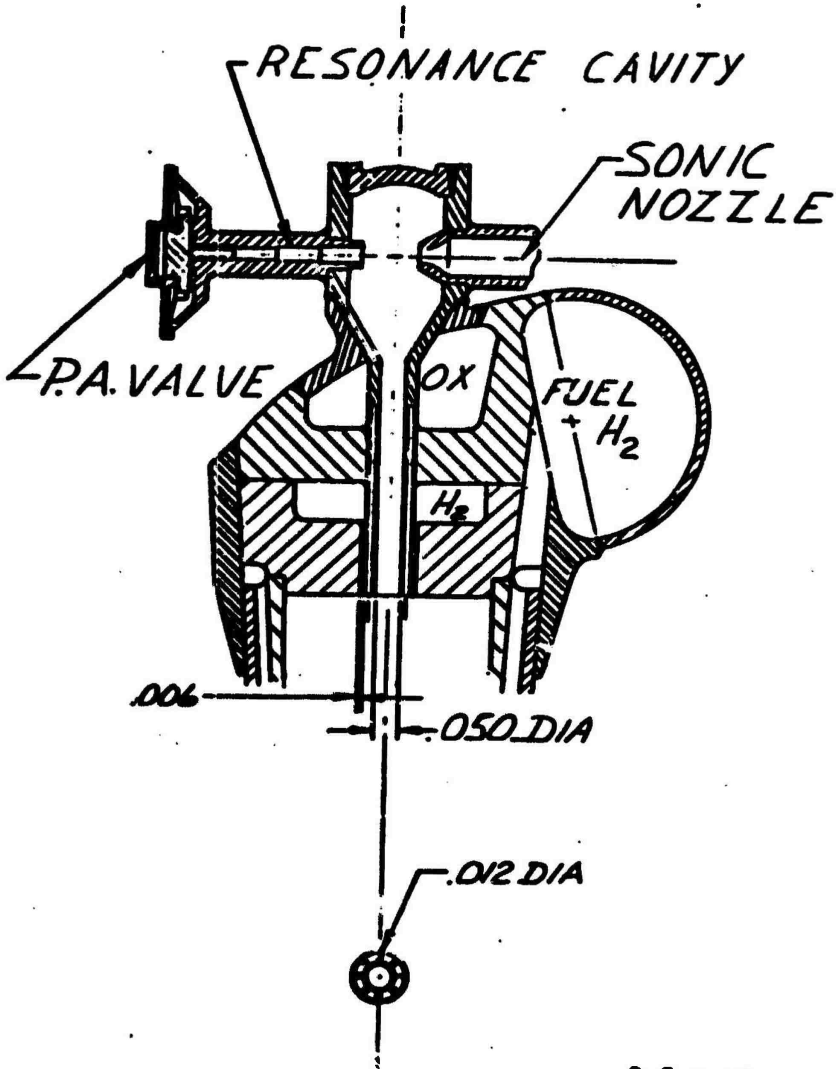
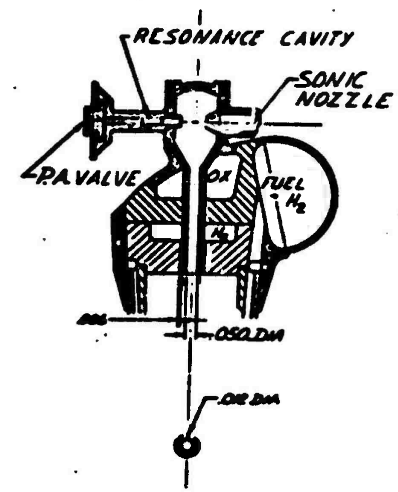


Figure D-4. Resonant Igniter



2.5 x size



Full Size

Figure D-5. Resonance Igniter

it is heated to over 2000 F. Pressure from the oxygen bottle actuates the pressure-actuated valve, permitting oxygen to mix with the hot hydrogen. The resulting combustion front is forced from the cavity by the O_2 flow. Combustion is sustained in the body by the opposed doublet of O_2 from the cavity and H_2 from the nozzle.

The resonance igniter design requires gaseous storage bottles for this configuration and would operate in a blowdown mode. Scaling from present designs, the flowrate was set at 0.001 lb/sec, or 5 percent of injector flow at ignition. Because the igniter bottles would be shut off during mainstage flow, this should cause no maldistribution in the injector. Because of the obvious installation ease for the tricentric element, it also was incorporated for the resonance igniter. The element would be sized to provide nominal mixture ratio at mainstage, with only H_2 flowing through the igniter. With the tricentric element, the center tube is part of the igniter. The outer tube extends from the structure between the O_2 and H_2 manifolds and is sealed on both sides. The outer annulus is formed by the injector face.

COMBUSTION WAVE IGNITER

The combustion wave igniter concept utilizes a spark-induced combustion wave passing through an unburned, gaseous oxygen/hydrogen mixture to ignite a pilot element at the main injector face. The combustion wave is initiated by an electrical arc discharge spark in a premix chamber. The resultant combustion wave begins to propagate in the unburned mixture in the direction of flow. Compression, shock, and eventually a detonation wave developes in the unburned mixture. A flow schematic for this ignition system is shown in Fig. D-6.

As presently conceived, the combustion wave ignition element is a set of triaxial tubes that are flush-mounted in the main injector face as shown in Fig. D-7. The core of the triaxial element is the combustion wave tube, and the annuli form the pilot element that is ignited by the combustion wave.

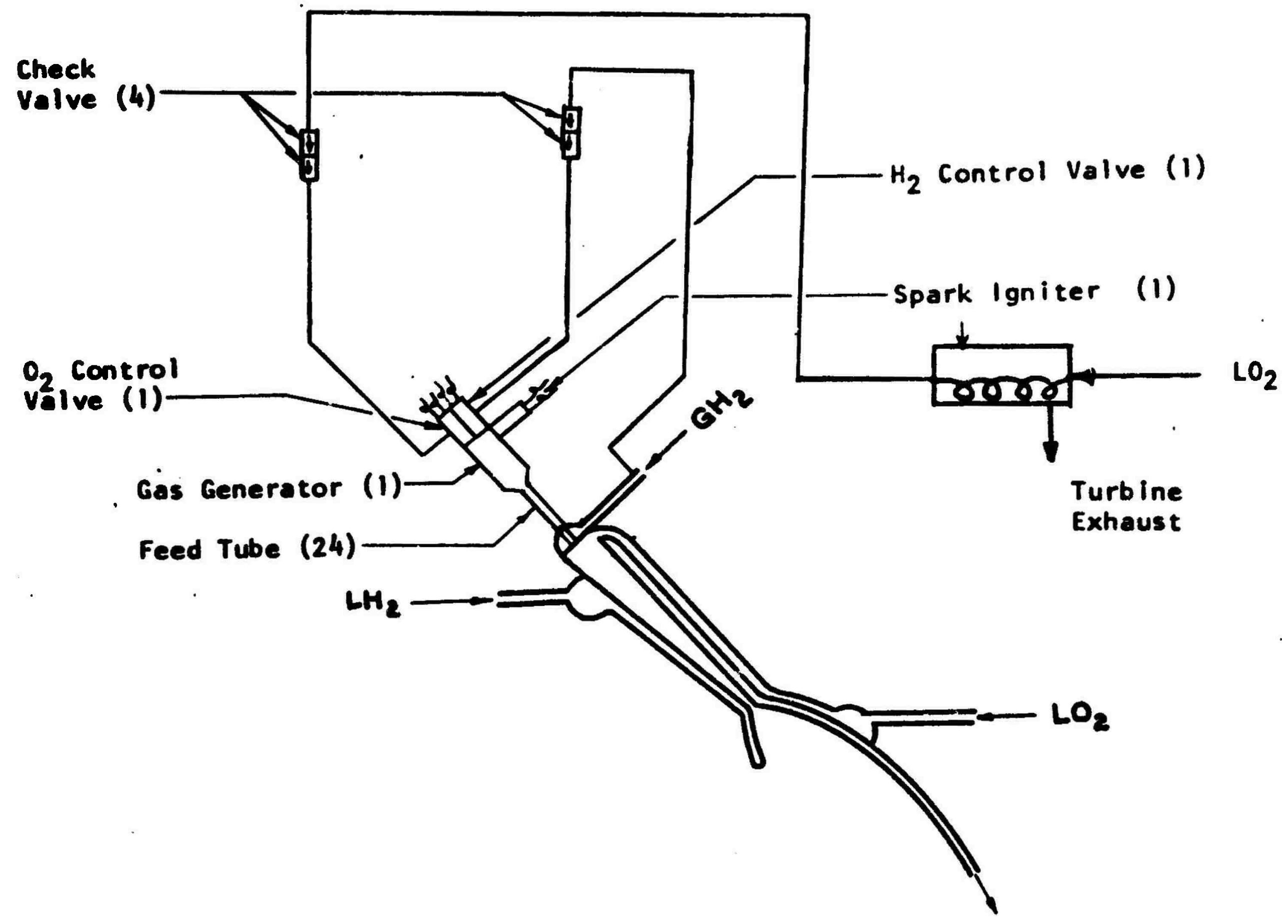


Figure D-6. Combustion Wave Igniter

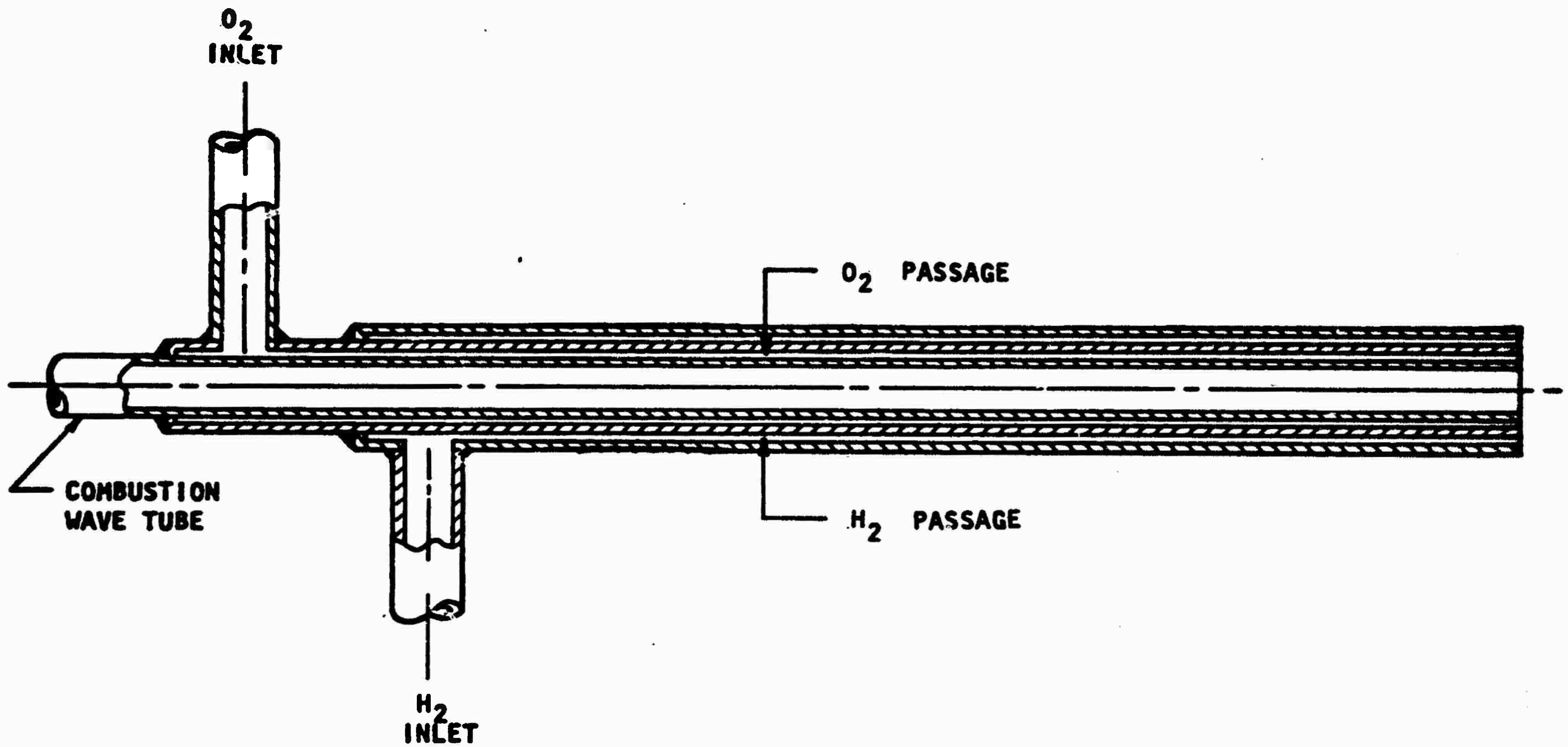


Figure D-7. Combustion Wave Igniter Element

The combustion wave for any number of these elements is supplied from a central premix chamber equipped with dual integrated spark plug/exciter units. A preliminary design of a premixer is shown in the component description section of this report. Its primary function is to prime the combustion-wave tubes with a combustible mixture.

The ignition energy is supplied to the premix chamber by redundant integrated plug/exciter units. The annular combustion wave manifold is supplied by the premix chamber and terminates in 24 triaxial elements, one for each chamber segment. Pilot flows in the element annuli around the combustion wave tube are supplied from the main injector manifolds.

Though present experience has not proved them necessary, refillable storage bottles may be required with the combustion wave system. The mixture ratio of the premixed propellants in the combustion wave tubes is critical to reliable operation of this system. This may be difficult to achieve because of the uncertainty associated with the flow conditions under tank-head operation. Storage bottles would eliminate this problem. Reliable operation of combustion wave ignition at less than 1 atmosphere has been demonstrated. Consequently, it is believed that combustion wave ignition under tank-head operation is considered feasible. If not, the ignition feed system would require the adoption of the storage bottles.

If bottles were required, the igniter oxidizer storage bottle would be filled with gaseous oxygen from the heat exchanger in the oxidizer turbine discharge. This heat exchanger also is necessary to provide GO_2 for main tank pressurization. The fuel bottle would be supplied with heated hydrogen after it passes through the turbines.

The sequence for the combustion wave ignition system is as follows:

1. At engine start signal, the engine main fuel valve is opened, followed by opening of the engine main oxidizer valve. The igniter element pilot manifolds are primed with propellants flowing to the thrust chamber.

2. The premix chamber valves are then opened and the combustion wave tubes are primeu.

3. Upon expiration of an ignition delay timer, the spark plug is fired and the premix chamber oxidizer valve is closed. The combustion wave propagates to the injector face and ignites the pilot flows in each segment.

The igniter elements are positioned at the centers of the segment injectors and are only slightly larger then the primary elements. During ignition, the igniter element mixture ratio will be 2, or greater, to ensure a hot enough flame to ignite the primary elements. During mainstage, the igniter element mixture ratio will be the same as the primary elements to minimize potential performance losses and prevent temperature profile maldistributions.

Related experience

A variety of experience has recently been accumulated with the combustion wave ignition concept. A series of 106 tests was conducted at the NR Los Angeles Division (LAD) Heat Transfer Laboratories, 142 tests at the Rocketdyne Research Area, and 13 tests at Rocketdyne CTL-3, all in support of the J-2 technology engine ignition system development. These experimental programs were conducted to: (1) evaluate the feasibility of generating a combustion wave in standard tubing of lengths typical of rocket engine systems, (2) map the pressure and mixture ratio limits of combustion wave generation, (3) evaluate the feasibility of igniting a pilot element with a combustion wave, (4) map pilot element ignition limits, (5) simulate proposed engine system valve sequencing,

(6) show the feasibility of multiple-element ignition, (7) demonstrate pilot element durability at simulated mainstage propellant conditions and pressures, and (8) demonstrate the feasibility of igniting the primary injector elements under simulated engine start conditions.

Combustion Wave Limits. During portions of the test program, the initial pressure and mixture ratio in the premix chamber were varied so that the combustion wave generation limits could be determined. The LAD test results, plotted in Fig. D-8, show that the lower limit of initial pressure was about 4.2 psia and the minimum mixture ratio for detonation was 2.3. The lower pressure limit agrees with the results of Ref. 7 in which spark-ignited detonations could not be consistently reproduced in hydrogen/oxygen mixtures at initial pressures below 1/4 atmosphere. The lower mixture ratio limit is in agreement with the induction length versus mixture ratio results in Ref. which indicates a lower mixture ratio limit of approximately 2.5 for propagation of a detonation wave. The research test program results, although not completely reduced and analyzed, tend to support the 2.3 minimum mixture ratio limit.

Pilot Ignition Limits. An investigation of pilot ignition limits was conducted and a pilot ignition map of the LAD tests is presented in Fig. D-9. The data indicate the minimum combustion chamber pressure for successful pilot element ignition decreases as mixture ratio is increased. No pilot ignitions were obtained below a combustion chamber pressure of 3 psia. The LAD test setup required a combustion chamber surrounding the igniter element tip to control the environmental pressure; therefore, it is possible that the pilot ignition limits shown in Fig. D-9 were influenced by the design of the backpressure device. Also, the igniter pilot element design may influence the ignition limits.

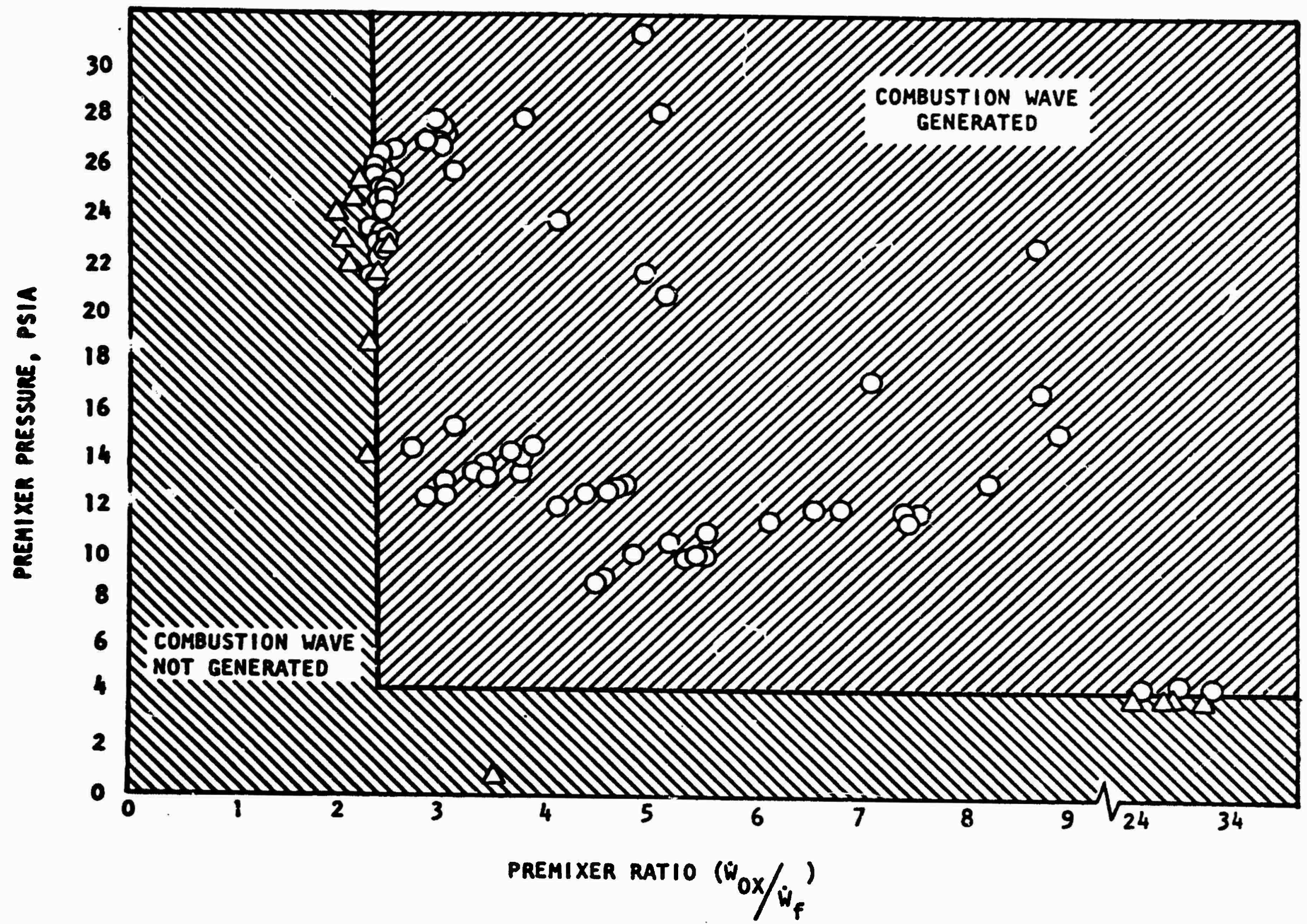


Figure D-8. Combustion Wave Map

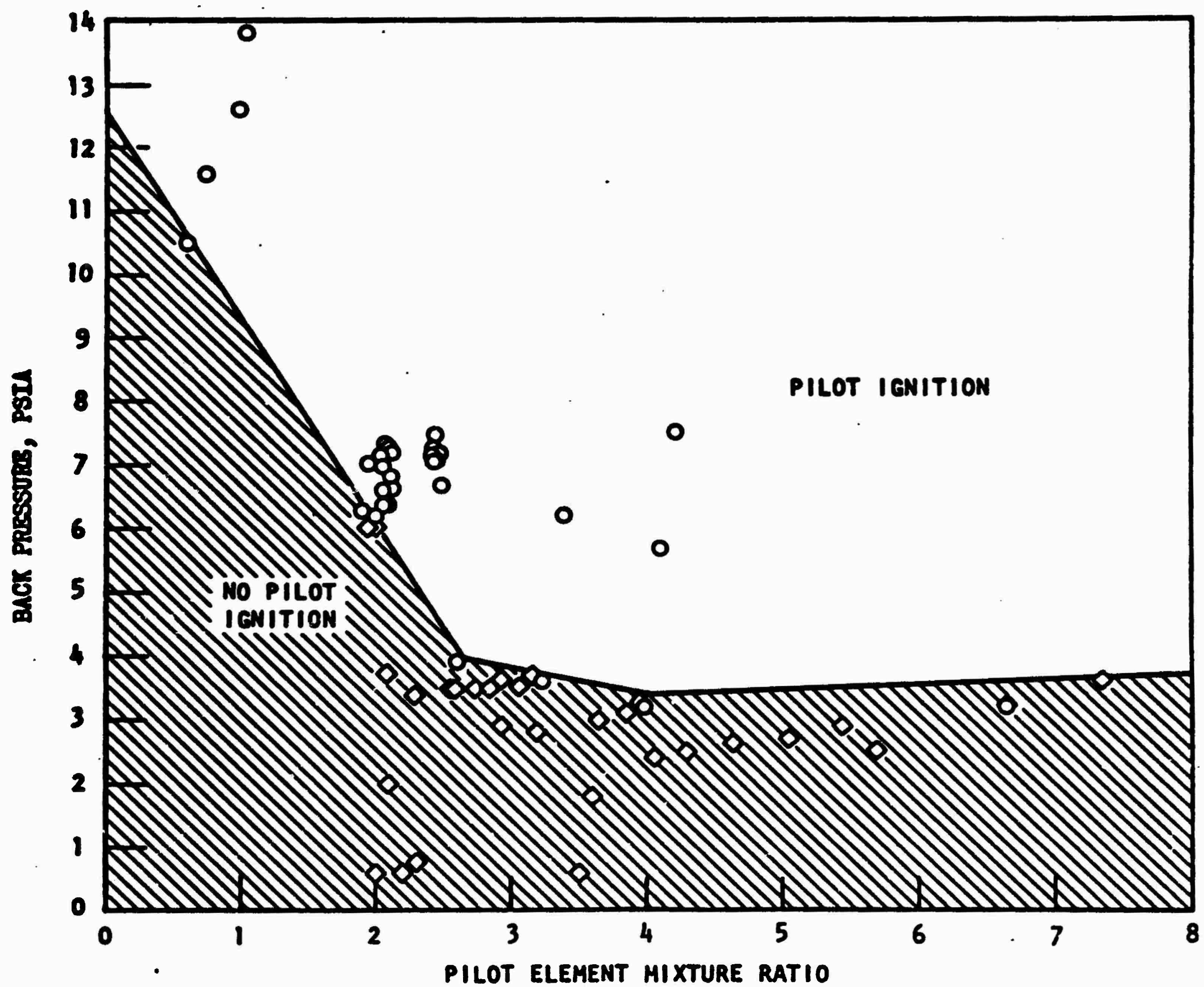


Figure D-9. Pilot Ignition Map

Valve Sequencing. The sequencing of the premixer oxidizer valve was investigated and the test results show that a combustion wave could not be generated if the oxidizer valve was closed prior to the spark signal. Delaying the oxidizer valve between 5 and 55 milliseconds from the spark signal resulted in no hardware damage and consistent combustion wave generation. These results led to the potential need for a check valve in the premixer oxidizer line to prevent hot-gas backflow into the igniter oxidizer supply line and to eliminate concern about the effect of critical valve timing on hardware integrity. Longer time delays may be possible, although they were not investigated.

Multi-element Tests. Many of the tests were conducted with multiple igniter elements (2, 3, 10, and 20) attached to the premix chamber. Successful pilot ignitions were obtained and ignition was successfully sustained, essentially within the same limits as those established for single-element tests.

Element Durability. A short series of tests was conducted to evaluate igniter element durability at mainstage flowrates, propellant temperatures, and chamber pressure. Element durability was satisfactory during these tests, although of short duration and somewhat below the linear thrust chamber breadboard engine mainstage chamber pressure of 1225 psia.

Propagation Tests. A series of 13 tests was conducted which successfully demonstrated feasibility of the combustion wave igniter to ignite the main injector elements under tank-head propellant inlet conditions. Simultaneous multisegment ignition (three linear thrust chamber segments) also was demonstrated during this test series.

Rechargeable Gaseous Propellant Supply Requirements

In the resonant, and possibly the combustion wave systems, rechargeable tankage sufficient for 1 second of operation was assumed. Start tank volume was governed by the case in which recharging occurs during a minimum-thrust firing, and start tank pressure was governed by the case in which recharging occurs during a full-thrust firing.

Two alternative techniques were evaluated: tank blowdown and regulated, uniform discharge. Based on the assumptions stated in Table D-4, the weights for each system were close to equal with the regulator weight equalling the tank weight differences for the two systems. As shown in Fig. D-10, the blowdown tanks provide 1 second of operation when charged initially to 200 psia. For an initial charge pressure of 1200 psia, the ignition source is adequate (i.e., $\dot{W}_{total} > 0.006$) for 3.4 seconds.

TABLE D-4. ASSUMPTIONS FOR TANK WEIGHT ESTIMATES

Gas Storage Pressure, psia	200 (min); 1200 (max)
Gas Storage Temperature, R	600 (GH_2); 340 (O_2)
Gas Flowrate (total at MR=1), lb/sec	0.006 (min); 0.008 (nom)
Tank Material	
Yield Stress, psi	100,000
Safety Factor	1.5
Density, $lb/in.^3$	0.28
Igniter Operating Duration, seconds	1.0 (min)

SYSTEM SELECTION

Based on the previously stated ground rules, only a few concepts remained as ignition system candidates, i.e., those just discussed. Final selection of the ignition system for the aerospike engine system was influenced primarily by the following criteria: (1) technology status, (2) design complexity, and (3) ignition system weight.

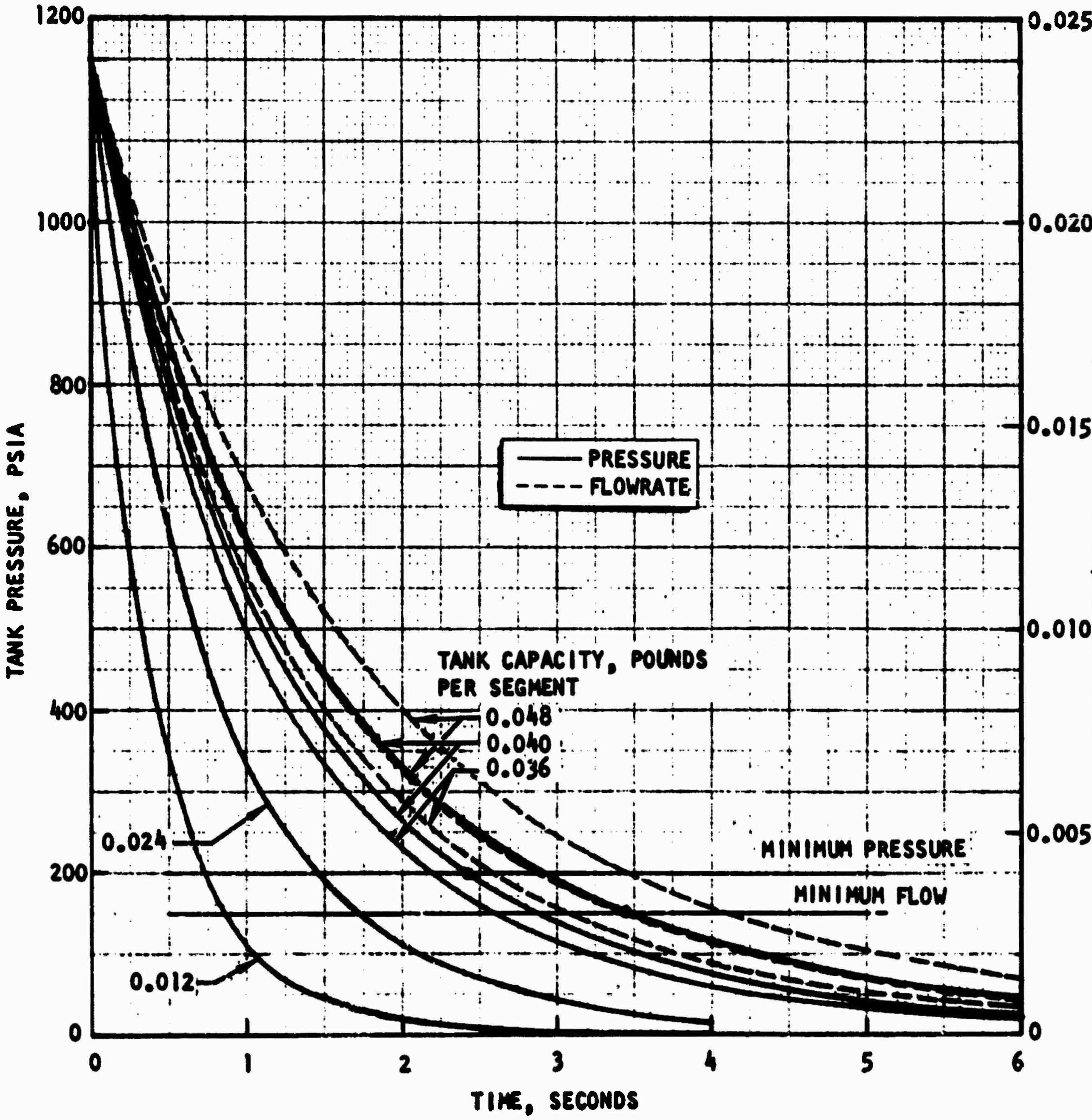


Figure D-10. Igniter Propellant Accumulator Blowdown Characteristics

As a result of this study, the combustion wave ignition system is recommended for the AMPT aerospike engine. A weight comparison of the candidate ignition system is shown in Table D-5. The combustion wave system without storage bottles is the lightest. Even with storage bottles its weight is comparable to a resonant system or a redundant spark system. The spark system poses the greatest problems with respect to integration into the thrust chamber/injector assembly for the AMPT aerospike application. Because of this design complexity and its heavier weight, the ASI system was eliminated. The technology status of resonance igniters has not been advanced to the extent of that of combustion wave systems, and they were eliminated for this reason, as well as for their heavier weight. Combustion wave ignition was selected because of the following reasons: (1) it is easily integrated into the thrust chamber assembly, (2) the possibility of utilizing tank-head-supplied igniter flows result in significantly lower weight than the other systems, and (3) operation at ignition and mainstage conditions applicable to the AMPT aerospike engine has been demonstrated.

TABLE D-5. IGNITION SYSTEM WEIGHT SUMMARY

Resonant System	Weight, pounds	Combustion** Wave System	Weight, pounds	Spark** Ignition System	Weight, pounds
Inlet Check Valves, Oxidizer (2)	0.5	Inlet Check Valves, Oxidizer (2)	0.5	Spark Igniters (24)	24.0
Inlet Check Valves, Fuel (2)	0.5	Inlet Check Valves, Fuel (2)	0.5	Wiring	1.5
Tank, Oxidizer	3.3				
Tank, Fuel	38.2				
Discharge Valve Oxidizer	0.5	Discharge Valve, Oxidizer	0.5		
Discharge Valve, Fuel	0.5	Discharge Valve, Fuel	0.5		
Gas Delivery Tubes	4.0	Gas Generator	0.5		
Resonators (24)	6.0	Gas Generator	0.5		
		Gas Delivery Tubes	2.0		
		Spark Igniters (2)	2.0		
Total	53.5	Total	6.5	Total	25.5

* If storage bottles are required to prime the premix chamber and combustion wave tubes, 41.5 pounds would be added for these bottles to the weight of this system.

** If redundant igniters are required to provide reliable ignition, the weight of this system will double.

Unclassified

Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1 ORIGINATING ACTIVITY (Corporate author) ROCKETDYNE a division of North American Rockwell Corporation 6633 Canoga Avenue, Canoga Park, California 91304	2a. REPORT SECURITY CLASSIFICATION Unclassified
	2b. GROUP

3 REPORT TITLE

O_2/H_2 Advanced Maneuvering Propulsion Technology Program Engine System Studies Final Report, Volume I: Aerospike Engine Configuration Design and Analysis

4 DESCRIPTIVE NOTES (Type of report and inclusive dates)

Final Report

5 AUTHOR(S) (First name, middle initial, last name)

Rocketdyne

6 REPORT DATE December 1971	7a. TOTAL NO OF PAGES ~~554~~ 575	7b. NO OF REFS 8
8a CONTRACT OR GRANT NO F04611-67-C-0116	9a. ORIGINATOR'S REPORT NUMBER(S) R-8807 (Volume I)	
b. PROJECT NO		
c.	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)	
d	AFRPL-TR-72-4	

10 DISTRIBUTION STATEMENT

Distribution limited to U.S. Government Agencies only; data based on test and evaluation; December 1971. Other requests for this document must be referred to AFRPL (STINFO), Edwards, California 93523.

11 SUPPLEMENTARY NOTES	12 SPONSORING MILITARY ACTIVITY Air Force Rocket Propulsion Laboratory Air Force Systems Command, USAF, Edwards AFB, California 93523

13 ABSTRACT

The engine system design and analysis studies provide a detailed definition of two 25,000-pound-thrust O_2/H_2 aerospike engines. The single-panel aerospike engine design point corresponds to the demonstrator thrust chamber configuration, specifically, chamber pressure and area ratio equal to 750 psia and 110:1, respectively. A second engine system and component design and operational description also is provided for the selected optimum aerospike engine employing a double-panel thrust chamber cooling circuit. The double-panel aerospike engine design has a chamber pressure and area ratio of 1000 psia and 200:1, respectively. These engine systems are designed to provide 5:1 throttling and off-design mixture ratio operation. The study effort also included the effects of variations in certain design parameters on engine performance, weight, propellant flow balances, life capability, development time and cost, and maintenance requirements. Additional parametric information is provided for design thrust levels between 8000 and 50,000 pounds.

DD FORM 1 NOV 65 1473

Unclassified

Security Classification

Unclassified

Security Classification

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Turbine Drives						
Engine System						
Turbopumps						
Thrust Chambers						
O_2/H_2 Aerospike Engines						

Unclassified

Security Classification